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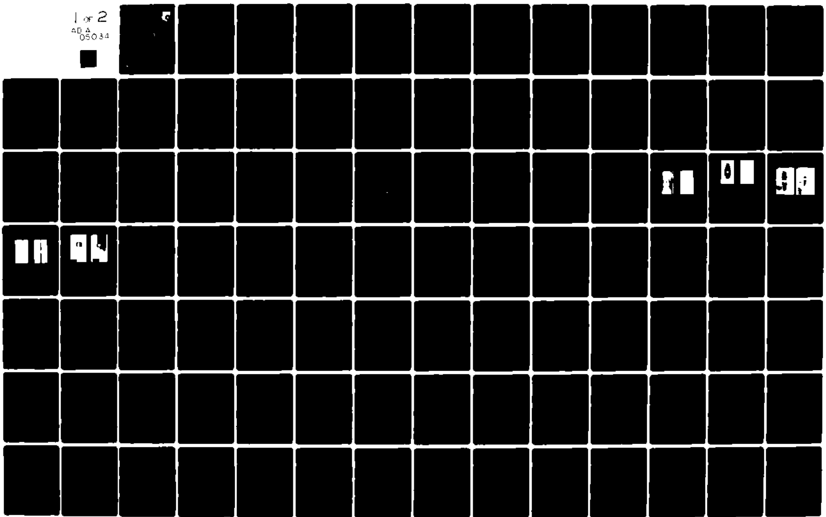

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EFFECTS OF FIGHTER ATTACK SPECTRUM ON COMPOSITE FATIGUE LIFE

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March 1981

TECHNICAL REPORT AFWAL-TR-81-3001
Final Report for Period September 1978 - October 1980

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This technical report has been reviewed and is approved for publication.

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The objective was to evaluate the effect of fighter wing load spectrum variations on the life behavior of composite structures. Six types of spectra were generated: (a) clipping to 90% test limit stress, (b) addition of stress overloads, (c) addition of low loads, (d) truncation to 70% test limit stress, (e) clipping of tension loads, (f) increased severity and number of air-to-air loads. A single hole compression test specimen was designed to simulate fatigue critical areas of fighter wing skins. Two different lay-ups:

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(a) fiber dominated (48/48/4), (b) matrix dominated (16/80/4), (% of 0° plies, % of $\pm$45° plies, and % of 90° plies) were selected. Both lay-ups were fabricated with 25 plies of .0104 inch nominal thickness Hercules AS/3501-6 graphite-epoxy prepregs. The spectrum fatigue life for each variation was predicted using constant amplitude fatigue data, a correlation parameter based on the concept of strain energy density factor for micro-cracks in the laminate matrix, an empirical modification to account for stress ratio effects on life, and a linear residual strength reduction fatigue damage model. Thirty-six constant amplitude and 184 spectrum tests were performed. Variations found to have greatest effect were those that increased the frequency or magnitude of high loads in the spectrum. Addition of overloads, and increased severity and number of air-to-air loads caused more than an order of magnitude reduction in life. A 5% increase in test limit stress caused 60% decrease in life. Variations found to have smaller impact on life were those that change the lesser loads in the spectrum. These variations are the addition of low-loads, truncation to 70% test limit stress, and clipping of tension loads. Recommendations and guidelines for deriving design and test spectra for multi-mission fighter aircraft were developed.

FOREWORD

This report was prepared by McDonnell Aircraft Company (MCAIR), St. Louis, Missouri, for the Structural Integrity Branch, Structures and Dynamics Division, Air Force Wright Aeronautical Laboratories, Wright-Patterson Air Force Base, Ohio, under Contract F33615-78-C-3218, Project 2401, Work Unit 24010125, "Design Spectrum Development and Guidelines Handbook". The contract was administered by John M. Potter, Project Engineer, AFWAL/FIBEC.

The Structural Research Department of McDonnell Aircraft Company was responsible for performance of this program. Principal Investigators were R. Badaliance and H. D. Dill. H. T. Young assisted in computer program operation. F. D. Coffey and B. C. Wilson, MCAIR Materials Laboratories, performed the test program.

This report covers work accomplished during the period September 1978 - October 1980.

This report was released by the authors in December of 1980 for publication.

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SECTION I

INTRODUCTION

Early design of aircraft structures concentrated primarily on the strength necessary to carry the loads induced in flight, and durability was not considered. As it has become important for airframe structures to be used reliably for many years under a variety of conditions, it has become necessary to verify both the strength and the durability of the airframe in ground test conditions.

The loading environment considered necessary to properly evaluate the aircraft of today has been developed from a long period of experience with aircraft behavior under a wide variety of conditions. The primary components of a loading spectrum consist of those loading elements that are of significant load level to cause damage. Only recently has it been necessary to decide whether and to what degree must a loading component be included in a loading spectrum because of its abnormal affect on the life behavior of the structure. For instance, in metallic structures the phenomena of crack growth retardation has caused difficulty in deciding the levels of the highest loads that will be allowed in a loading environment to properly test the fleet durability of a representative airframe. The retardation phenomena causes an abnormal change in fatigue behavior in that a marked increase in fatigue life results from the addition of one or more loads of a level only slightly larger in magnitude than the largest in the baseline spectrum. This abnormality in fatigue behavior forces the individual developing a design or test load spectrum for a metallic structure to make a decision as to the level and number of the highest loads to be applied in order to obtain a representative test result.

The behavior of composite laminates under fatigue loading is quite different from that of metallic materials. The difference is that in composite materials there is no single dominant flaw which grows and leads to eventual failure. Since failure mechanisms in composites are different from the metals then it is possible that the loading components causing any abnormal behavior will be different and any special design and verification test load spectrum requirements will be also different.

While the phenomenon of retardation is now well-known in metallic structural fatigue there have been no systematic studies of spectrum sensitivities in structures manufactured from composite materials. The purpose of this effort was to determine the existance of any abnormal load spectrum related fatigue life effects in composite structures. In order to determine the existance of these effects a series of tests were run to determine the relative sensitivity of composite structures to variations in fighter aircraft load spectra that were chosen as possibly being extraordinarily significant to this material. As a result of the study, recommendations were developed for the creation of design and structural verification test spectra that would lead to a representative evaluation of a structure manufactured from this type material. The study was performed in five phases.

In Phase 1, Load Sequence Generation, eleven spectra were generated which were derived from four baseline load factor spectra for the F-15 aircraft. These baseline spectra are Air-to-Air; Air-to-Ground; Instrumentation-and-Navigation; and a combination of these, the F-15 Measured Mix-Truncated. The upper wing skin of the F-15 aircraft was used as a basis to convert load factor to stress. Cycle-by-cycle stress histories were generated for the baseline spectra and modified to create the spectra variations.

The spectra variation types considered were: (a) Clipping to 90% test limit stress, (b) Addition of 115% test limit stress overloads, (c) Addition of 125% test limit stress overloads, (d) Addition of low loads to match original measured mix, (e) Truncation to 70% test limit stress, (f) Clipping of tension loads, (g) Increased severity and number of air-to-air loads. The cycle-by-cycle stress histories were generated using a computer program described in Reference 1. The modifications of these random time histories used to create the spectra variations were developed with another computer program, described in Reference 2.

In Phase II, Test Specimen Design and Manufacture, a simple compression test coupon was selected for manufacture and test. Two lay-ups were selected: Fiber dominated and matrix dominated. Fiber dominated lay-ups (laminates with a high percentage of 0° plies) are representative of wing torque box skin sreas where the design is strength controlled. Matrix dominated lay-ups (laminates with a high percentage of $\pm$ 45° plies) are used in stability critical structural components such as fixed trailing edges. More than 60 specimens of each laminate type were manufactured, and subsequently tested.

In Phase III, Analytical Life Predictions, the spectrum fatigue life for each of the variations was predicted. These predictions were made using constant amplitude fatigue data, and a new correlation parameter which was developed based on the concept of strain energy density factor for micro-cracks developed in the laminate matrix. This methodology was used in conjunction with a linear fatigue damage model to predict spectrum life.

In Phase IV, Experimental Verification, 36 constant amplitude and 184 spectrum tests were performed. The purposes of the test program were to evaluate the effects of spectra variations, and provide data useful for defining guidelines for structural verification of future aircraft. All spectrum tests were performed using compression dominated load spectra typical of fighter upper wing skins.

In Phase V, Recommendations and Guidelines, the experimental data were evaluated and summarized, and recommendations and guidelines were developed for deriving design and test spectra for multi-mission fighter aircraft.

SECTION II

SPECTRUM GENERATION

1. LOAD FACTOR EXCEEDANCE CURVES - Load factor spectra were developed based on the F-15 aircraft, using exceedance data developed during the design phase of that aircraft, updated based on tracking of service usage. Cycle-by-cycle stress histories were created using procedures previously developed under the program, "Effects of Fighter Attack Spectrum on Crack Growth", Reference 1. These procedures are summarized in References 2 and 3; input data for the procedures are presented in Appendix A.

The planned operational usage for the F-15 included a specific set of missions each with its own requirements. There were four different air-to-air missions, two different air-to-ground missions, and one instrumentation and navigation training type of flight. Each of these missions was further subdivided into mission segments, viz., ascent, cruise, combat, descent, and loiter. A schematic of the F-15 load factor exceedance curves for these mission segments is shown in Figure 1 as peak exceedances per hour of time spent in a given segment. For each of the seven basic F-15 missions, the time spent in each mission segment was determined from mission analysis. The total load factor exceedance curves were obtained by summation of the exceedances for the seven missions. The expected usage of the aircraft, termed the F-15 Design Mix spectrum, is given in Figure 2 as load factor exceedances. It is based on 475 hours of air-to-air, 325 hours of air-to-ground, and 200 hours of instrumentation and navigation.

The F-15 Design Mix spectrum has been updated based on studies of F-15 operation usage data obtained from a loads monitoring program. Counting accelerometers are installed on every aircraft and signal data recorders are installed on 20% of the fleet and record 22 flight parameters from one to thirty times per second. The F-15 operational load factor exceedance curve, based on 60,000 hrs of measured aircraft usage, is termed the Measured Mix and is the baseline spectrum for this program. The Measured Mix and the Design Mix exceedance curves are very similar, as shown in Figure 2.

The F-15 exceedance curves are typical for multi-mission fighter aircraft, as demonstrated by comparisons to F-4 data. Load factor exceedance data from air-to-air missions studied during the F-4C/D and the F-4E(S) ASIP programs are compared to the F-15 air-to-air load factor exceedance curves in Figure 3. Because of the better maneuvering capability provided by the leading edge flaps on the F-4E(S) data tends to approach the F-15 air-to-air curve. Load factor exceedance data from air-to-ground missions also studied during the F-4 ASIP programs are compared to the F-15 air-to-ground load factor exceedance curves in Figure 4.

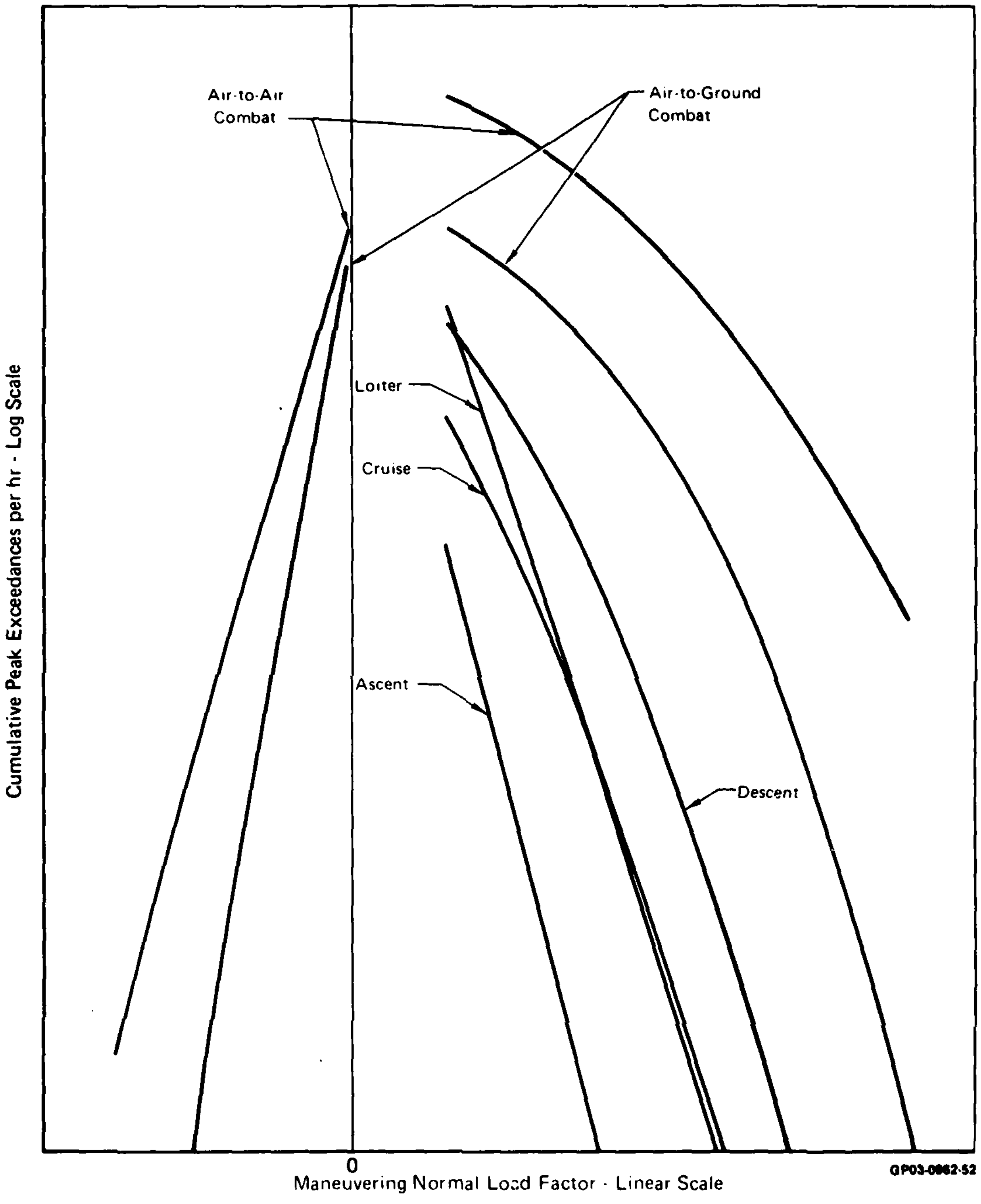

Figure 1. F-15 Mission Segment Exceedance Curves

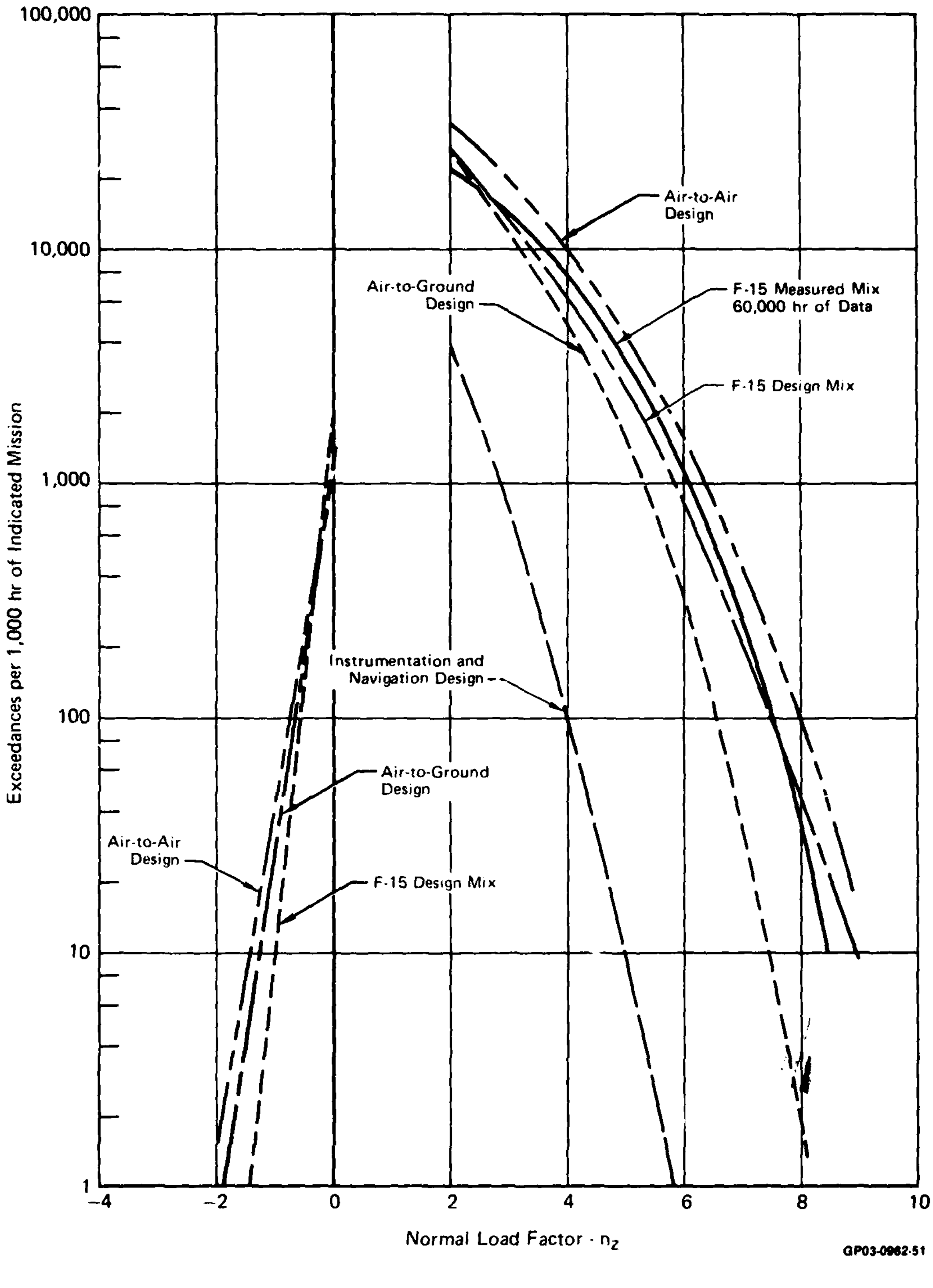

Figure 2. Baseline Load Factor Spectra

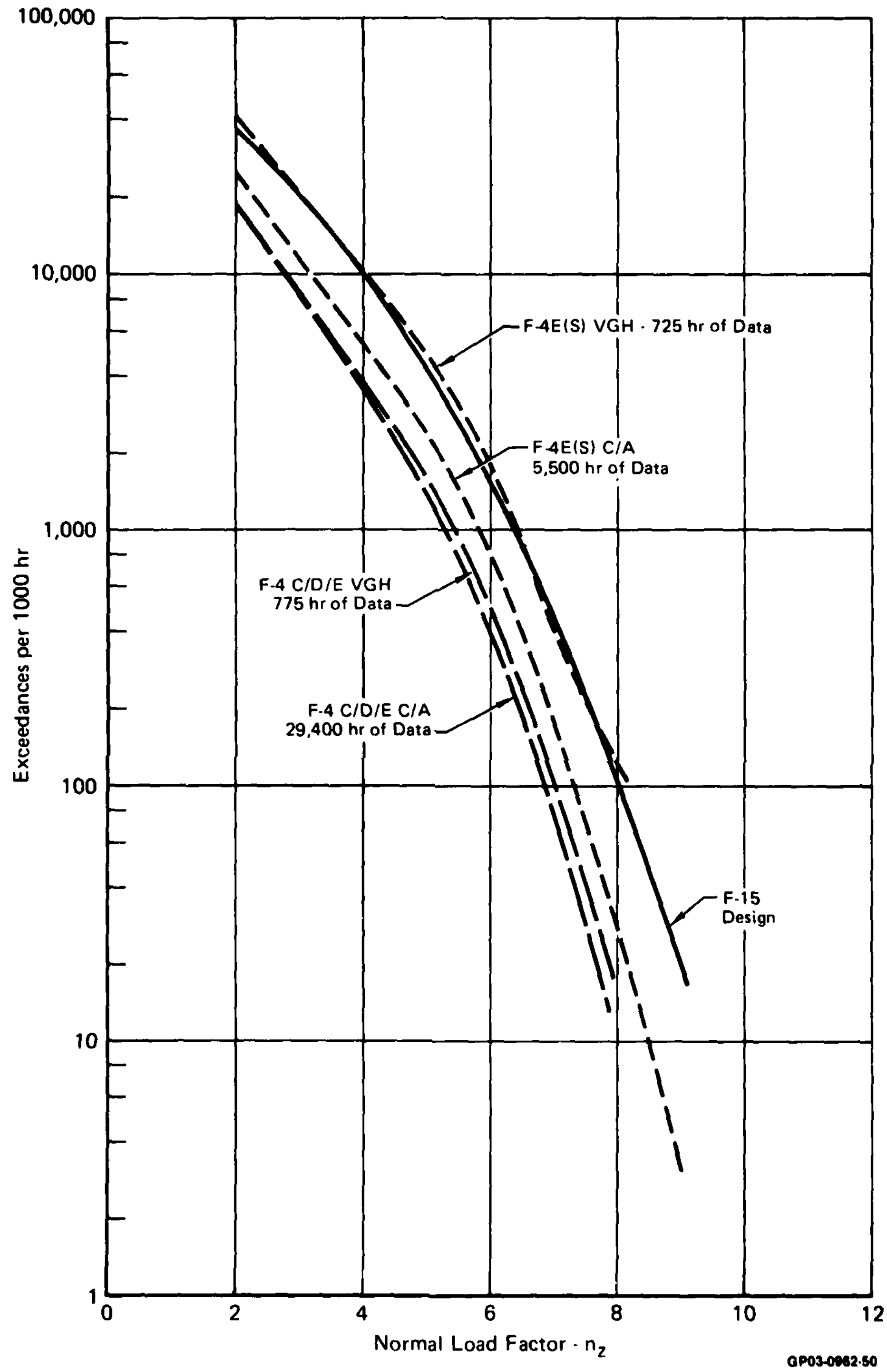

Figure 3. Comparison of Air-to-Air Exceedance Curves

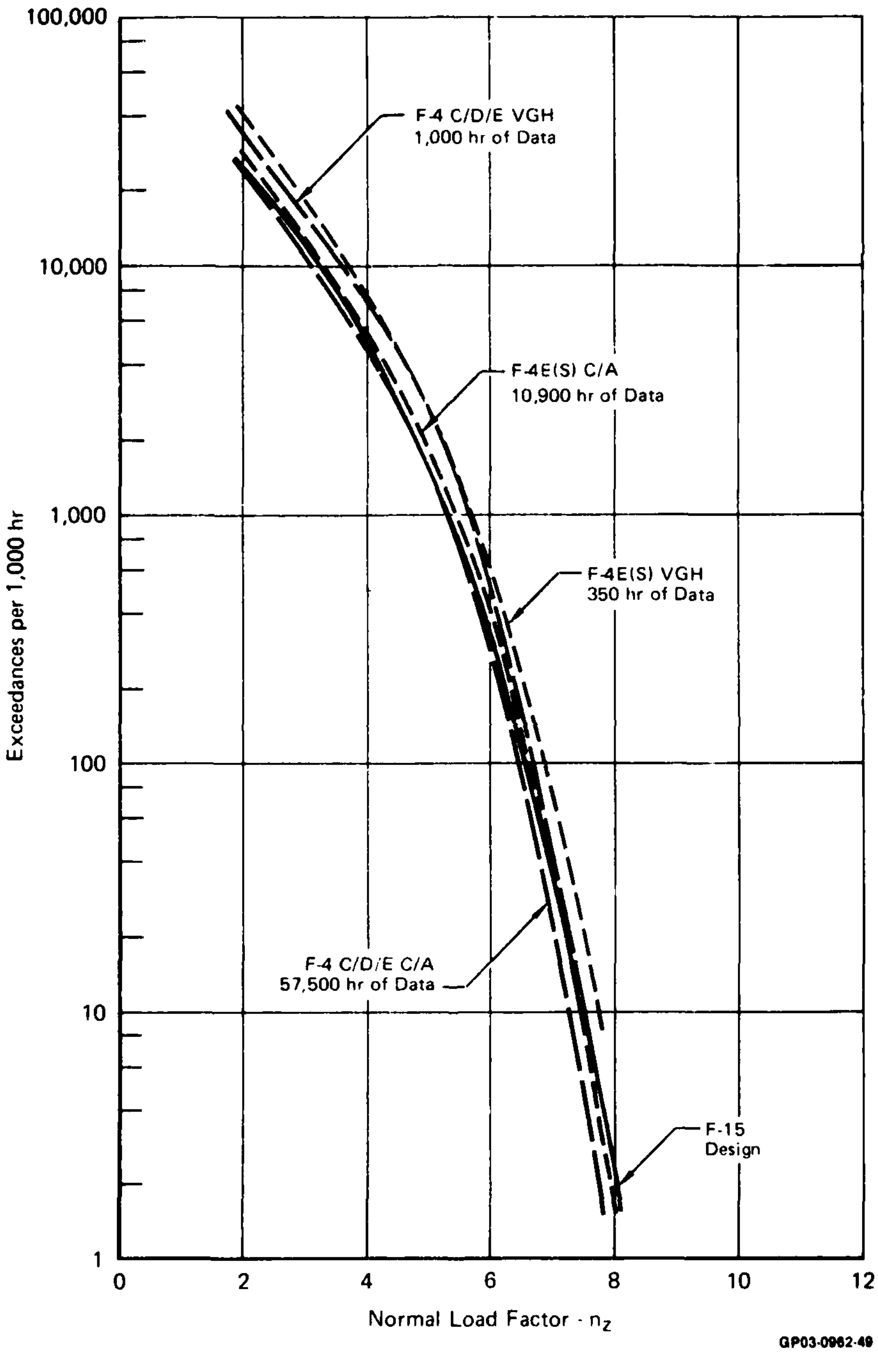

Figure 4. Comparison of Air-to-Ground Exceedance Curves

All three aircraft (F-15, F-4C/D, F-4E(S)) have fairly similar load factor exceedance curves for air-to-ground missions where maneuvering capabilities are not very important. The overall average curves for the F-4 C/D/E, F-4E(S), F15 and F-18 are shown in Figure 5, and their similarity is evident.

2. STRESS EXCEEDANCE CURVES - In order to generate a spectrum for analysis and test, it was first necessary to convert load factor to stress at critical locations. During the design of the F-15, external loading distributions for each flight condition were determined. Internal structural loads for these conditions were computed using finite element models. From these data, stresses at critical locations were correlated with normal load factor for the different flight conditions. A typical correlation for the upper wing skin is shown in Figure 6. These graphs were then used in conjunction with the load factor frequencies to develop the stress exceedance curves (Figure 7). The F-15 Measured Mix stress exceedance curve was based on the measured history of load factor, using the load factor to stress conversion derived from analyses performed during the F-15 design.

The F-15 Measured Mix stress exceedance curve was matched by combining cycle-by-cycle stress histories based on design Air-to-Air, Air-to-Ground, and Instrumentation-and-Navigation stress exceedance curves. This required a modest increase in percentage of Air-to-Air missions in the mix, as compared to the design mix. The distribution of hours and stress exceedances in the Design and Measured Mix Spectra are summarized in Table 1.

3. BASELINE SPECTRUM VARIATIONS - The load spectrum used was based on the projected stress history from the upper wing skin. This skin surface and its stress spectra was selected since composite structures are expected to be more susceptible to compression fatigue degradation than the tension skin. Because the upper wing skin location was selected, a positive structural "load factor" results in a negative or compressive applied load. The spectra variations evaluated through test are outlined in Tables 2 and 3. The test series for the fiber dominated lay-up (Table 2) and matrix dominated lay-up (Table 3) are nearly identical except for changes required by the extremely long fatigue lives exhibited by fiber dominated lay-ups. Because of these long lives, the test limit stress for the fiber dominated lay-up was 80% of F_{cu} for the test specimen, in contrast to 66.7% of F_{cu} for the matrix dominated lay-up. The use of this higher stress level for the fiber dominated lay-up reduced lives and resulted in reasonable test times. It limited the magnitudes of overloads and ranges of design limit stress that could be investigated.

a. Baseline Spectrum - The baseline spectrum is a cycle-by-cycle history based on the F-15 Measured Mix load factor exceedances. This spectrum was truncated at approximately 55% test limit stress, resulting in 5,000 stress cycles per thousand hours (Figure 8). This truncation was done for test economy. An evaluation of the effect of this truncation is included in Low Load Truncation variations, as outlined in Tables 2 and 3.

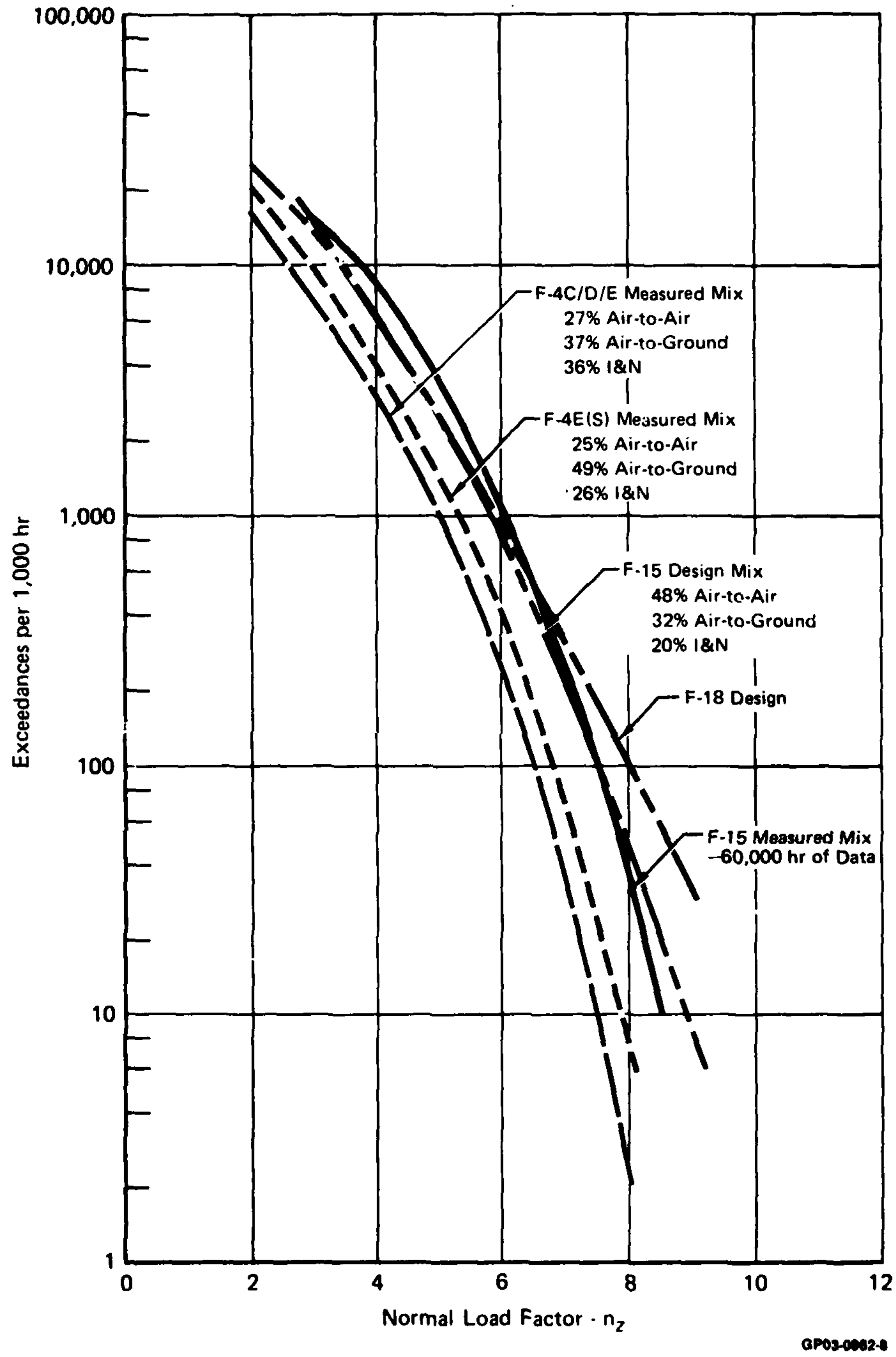

Figure 5. Comparison of Measured Mix Exceedance Curves

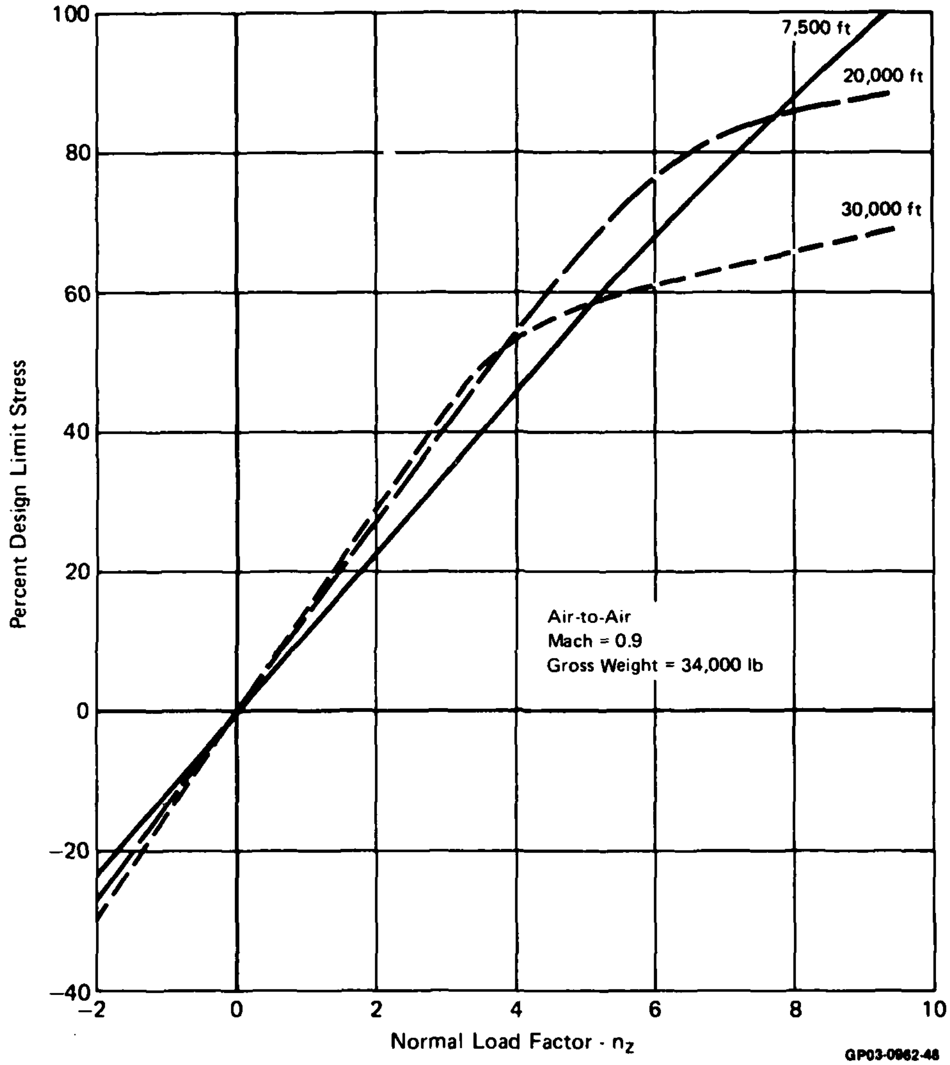

Figure 6. Load Factor to Stress Conversion

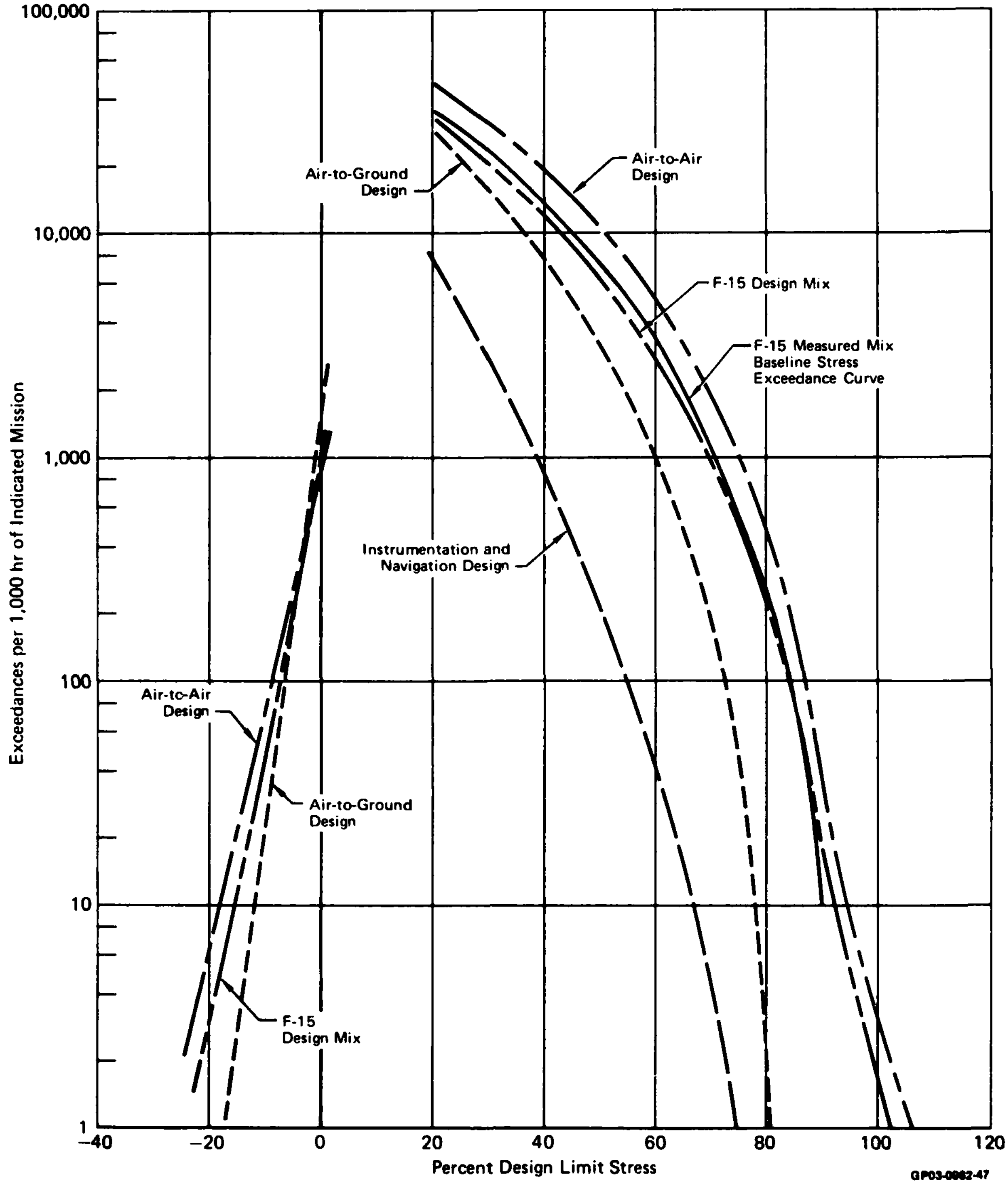

Figure 7. Baseline Stress Spectra

TABLE 1. DISTRIBUTION OF HOURS AND EXCEEDANCES IN DESIGN AND MEASURED MIX SPECTRA

	Design Mix		Measured Mix	
	Effects of Fighter Attack Spectrum on Crack Growth Reference 1.		Current Program	
	Hours	Exceedances of 60% Limit Stress	Hours	Exceedances of 60% Limit Stress
Air-to-Air	475	2,140	700	3,150
Air-to-Ground	325	450	100	140
Instrumentation and Navigation	200	10	200	10
Total	1,000	2,600	1,000	3,300

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TABLE 2. SPECTRA SUMMARY
Fiber Dominated Lay-Up

Spectra Type	Description	Number of Spectra
Baseline	1. F-15 Measured Mix-Truncated	1
High Load Clipping	1. Clipping to 90% Test Limit Stress 2. Addition of 115% Test Limit Stress Overloads	2
Low Load Truncation	1. Addition of Low Loads to Match Original Measured Mix 2. Truncation to 70% Test Limit Stress 3. Clipping of Tension Loads	3
Design Stress Level	1. 90% of F_{cu} 2. 80% of F_{cu}	2
Usage	1. Increased Severity and Number of Air-to-Air Loads 2. Air-to-Ground Baseline	2
Total		10

Note: Test limit stress = 85% F_{cu}

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TABLE 3. SPECTRA SUMMARY
Matrix Dominated Lay-Up

Spectra Type	Description	Number of Spectra
Baseline	1. F-15 Measured Mix-Truncated	1
High Load Clipping	1. Clipping to 90% Test Limit Stress 2. Addition of 125% Test Limit Stress Overloads	2
Low Load Truncation	1. Addition of Low Loads to Match Original Measured Mix 2. Truncation to 70% Test Limit Stress 3. Clipping of Tension Loads	3
Design Stress Level	1. 85% of F_{cu} 2. 55% of F_{cu}	2
Usage	1. Increased Severity and Number of Air-to-Air Loads 2. Air-to-Ground Baseline	2
Total		10

Note: Test limit stress = 66.7% F_{cu}

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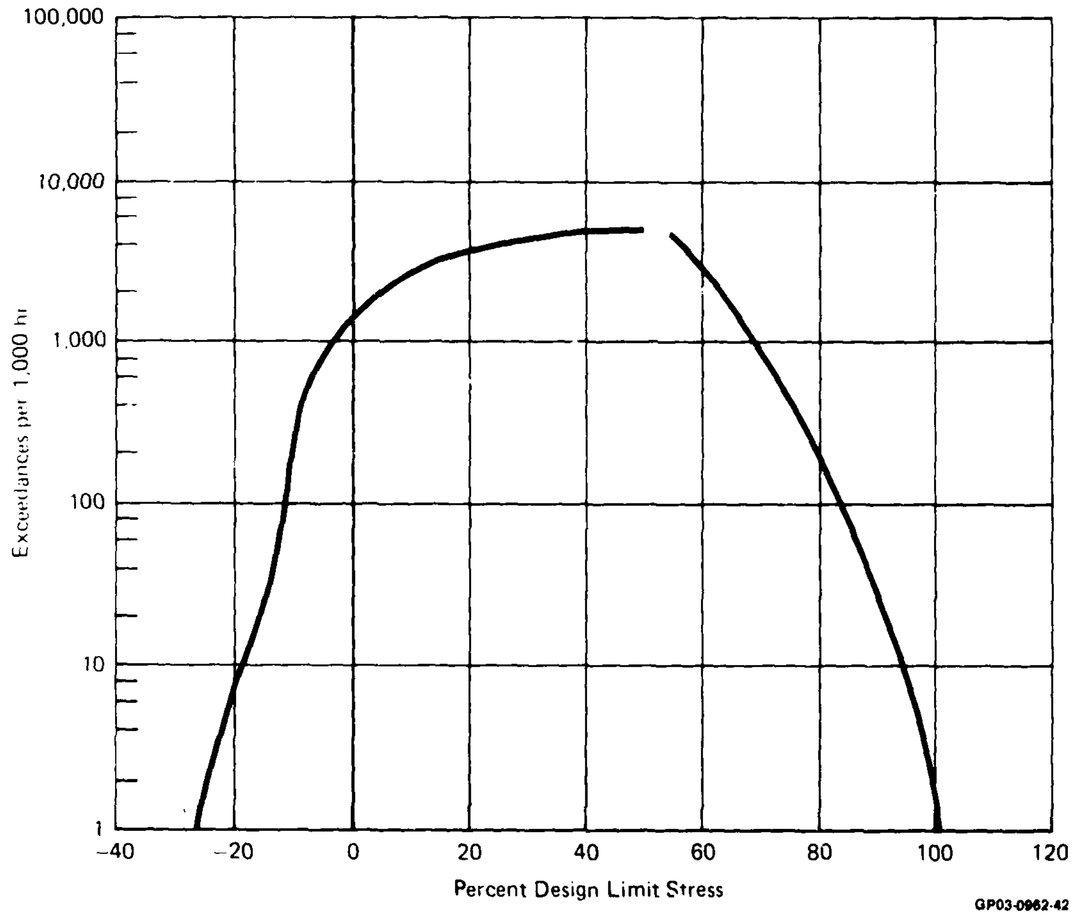


Figure 8. Baseline Spectrum: Measured Mix-Truncated

The cycle-by-cycle stress histories were developed using digital techniques based on random noise theory, as described in Reference 1. An advantage of this approach, using power spectral density (PSD), is that both the exceedance content and the frequency content of the history are retained. The preservation of the frequency in the analysis assures proper coupling of peaks and valleys. Stress histories developed using the PSD approach appear similar to measured histories.

The spectrum generation procedures are incorporated into two computer programs. The first program generates cycle-by-cycle stress histories for air-to-air, air-to-ground, and instrumentation and navigation baseline missions. Gaussian stress histories are generated, which have peak and valley exceedance curves that are symmetric, and have zero mean. These Gaussian histories are mapped into "real" histories whose exceedances are asymmetric with a non-zero mean. The theory for this program is described in Reference 1, and the program is documented in Reference 2. The input include PSD data, coefficients that are used to transform the Gaussian stress histories into "real" histories, and coefficients used to eliminate small stress excursions (rise and fall counting). These data are the same as used in the Reference 1 study. Input data are summarized in Appendix A. The second program combines the appropriate histories, inserts the ground loads between missions, and performs the data management operations required to develop the Design Mix baseline spectrum, and the spectra variations. This program is documented in Reference 3, input data are summarized in Appendix A.

b. Variations

(1) High Load Clipping - The addition of high loads (tension) to a spectrum when used to test metal structure usually results in increased fatigue life. Results of analyses described in Section IV, infer that high loads (compression) will be slightly damaging in composite structure. Conversely, the clipping of high loads in a spectrum used to test metal structure usually reduces life, whereas in composite structure clipping would be expected to increase life. This major difference in expected behavior between composite and metal structure merited evaluation. Hence, the high load clipping variations indicated in Tables 2 and 3 were tested. Exceedance curves are summarized in Figure 9.

(2) Low Load Truncation - For test economy, the baseline spectrum (F-15 Measured Mix - Truncated) was truncated to approximately 55% TLS which results in 5,000 load cycles per thousand hours. The analysis presented in Section IV.4 indicates stresses below that level are not significantly damaging, but that some damage is developed. A test spectrum was created by addition of low load levels to match the original Measured Mix spectrum, which contains approximately 18,000 load levels (Figure 10). To further determine the effects of load truncation, removing all loads below 70% TLS was used as a test spectrum. The basic load spectrum that was used is compression dominated. This spectrum was selected because composites are expected to be more susceptible to compression fatigue degradation, than tension. In addition to the variations described above, clipping the tension (valley) loads from the baseline spectrum was an additional low load truncation variation.

(3) Design Stress Level - The ability of predictive techniques to estimate fatigue life, and the effect of stress level on fatigue life, was investigated by systematically varying the test limit stress as indicated in Tables 2 and 3.

(4) Usage Variations - The most severe component of the baseline spectrum is the air-to-air combat segment. Changes in usage that cause differences in this component can be expected to have significant impact on life. Therefore, increasing both severity and number of air-to-air loads was used as a usage variation. Air-to-ground is a less severe component, and was also evaluated as a usage variation. Exceedance curves are presented in Figure 11.

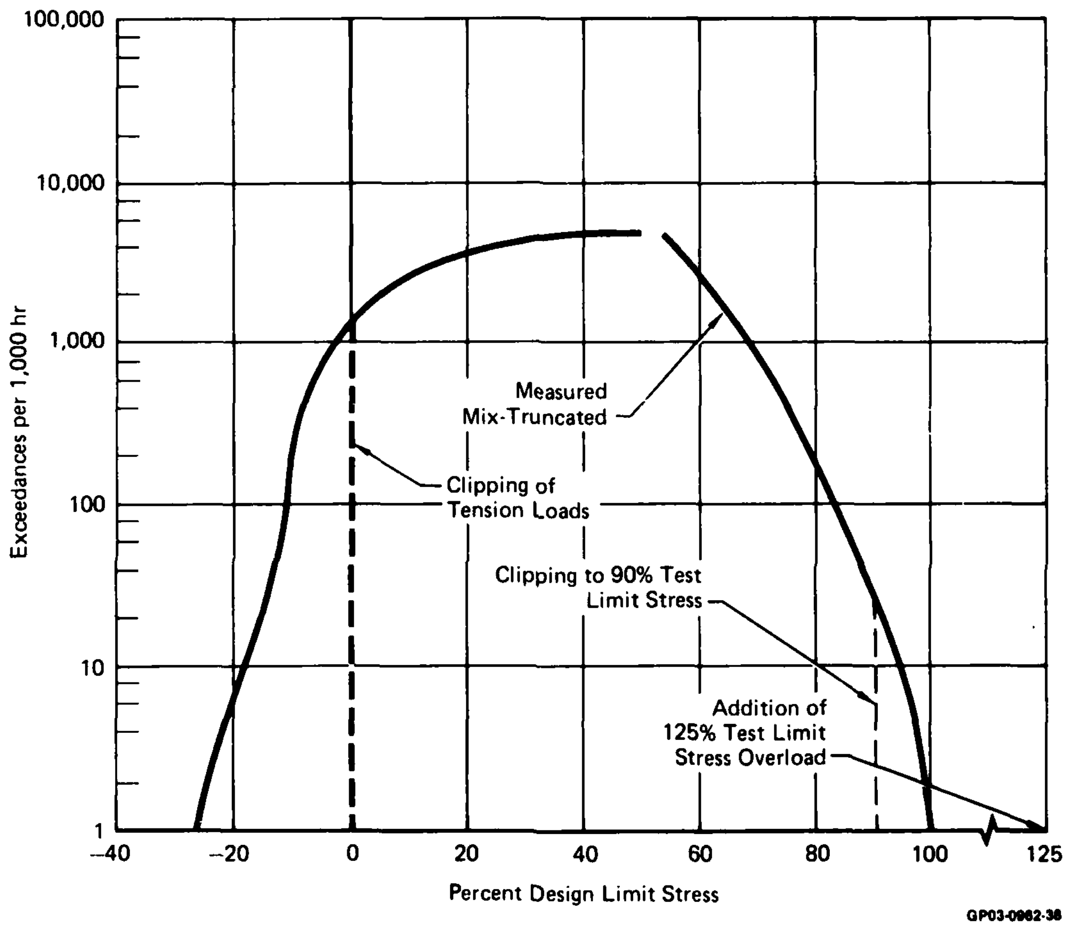


Figure 9. Exceedance Curves for Clipping Variations

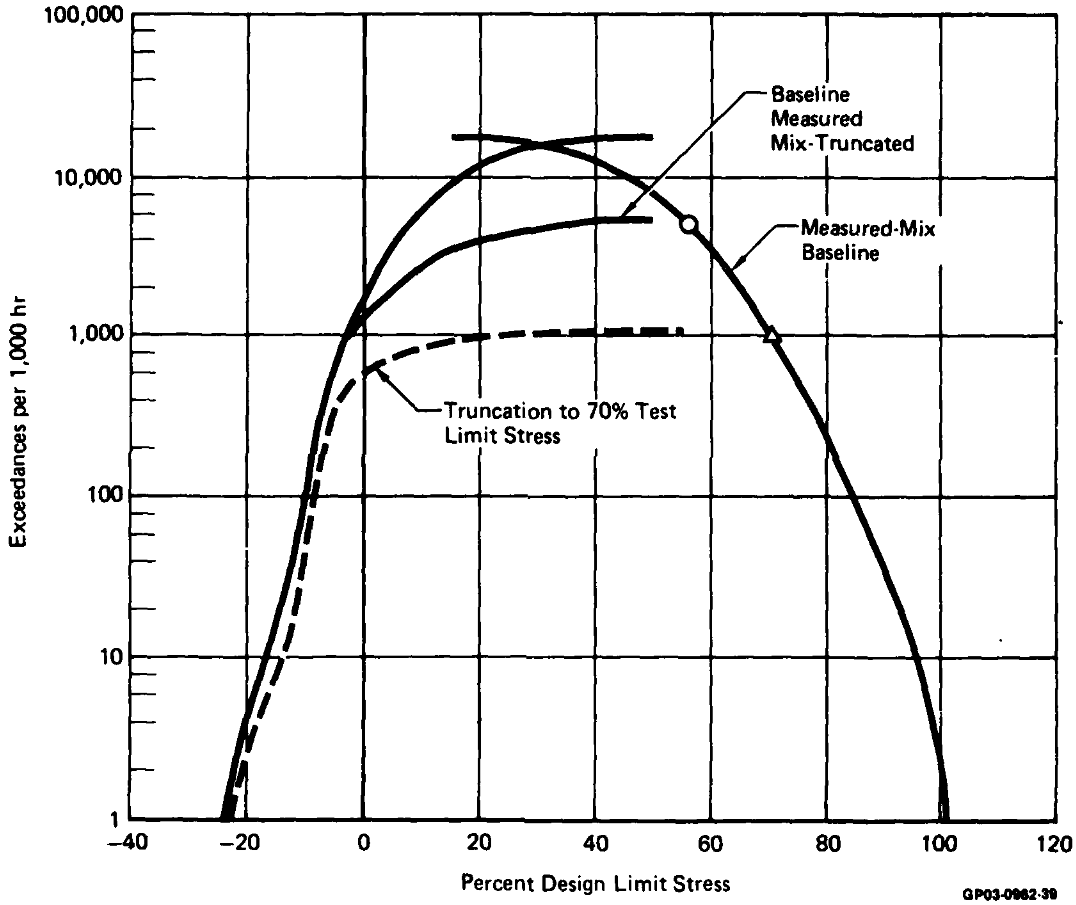


Figure 10. Exceedance Curves for Truncation Variations

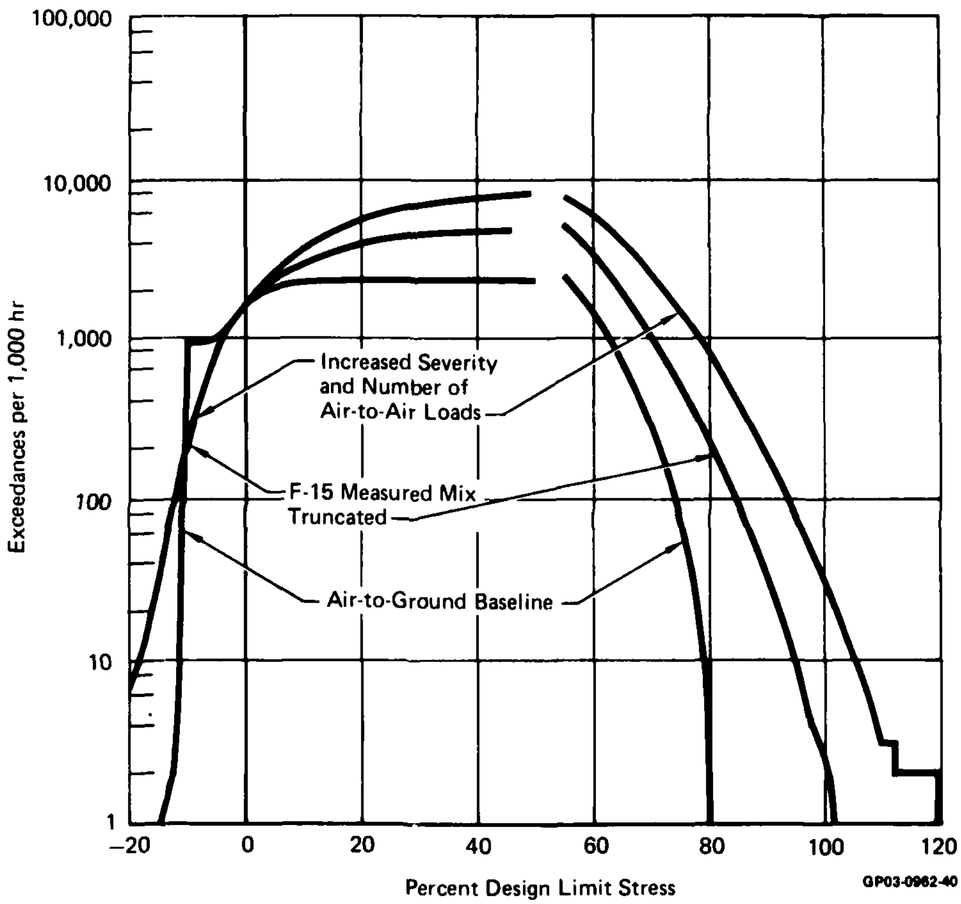


Figure 11. Exceedance Curves for Usage Variations

SECTION III

SPECIMEN DESIGN AND MANUFACTURE

1. SPECIMEN GEOMETRY - The single hole test specimen (Figure 12) was designed to simulate fatigue critical areas of fighter wing structure. Generally, structural elements susceptible to fatigue damage are those with holes or notches, and usually are associated with mechanically fastened joints. These joints in composite structures are similar to those seen in metal structures. The fatigue life of elements of these joints is influenced by the percent of load transferred, and by geometry. Low load transfer is typified by the attachment of a wing skin to a spar in a multi-spar wing, where the span-wise load in the skin is much greater than the load introduced locally by the fasteners. This joint approaches the case of an unloaded hole, in terms of bearing by-pass stress interactions. This type of structural element can be simulated by a specimen with a single or in-line fastener hole, that can incorporate geometric constraints such as edge-distance-to-hole diameter ratios.

The single hole test specimen shown in Figure 12 does not require tab ends. The stress concentration created by the fastener hole is the fatigue critical area. Only the volume of the laminate that is near the fastener hole is fatigue critical.

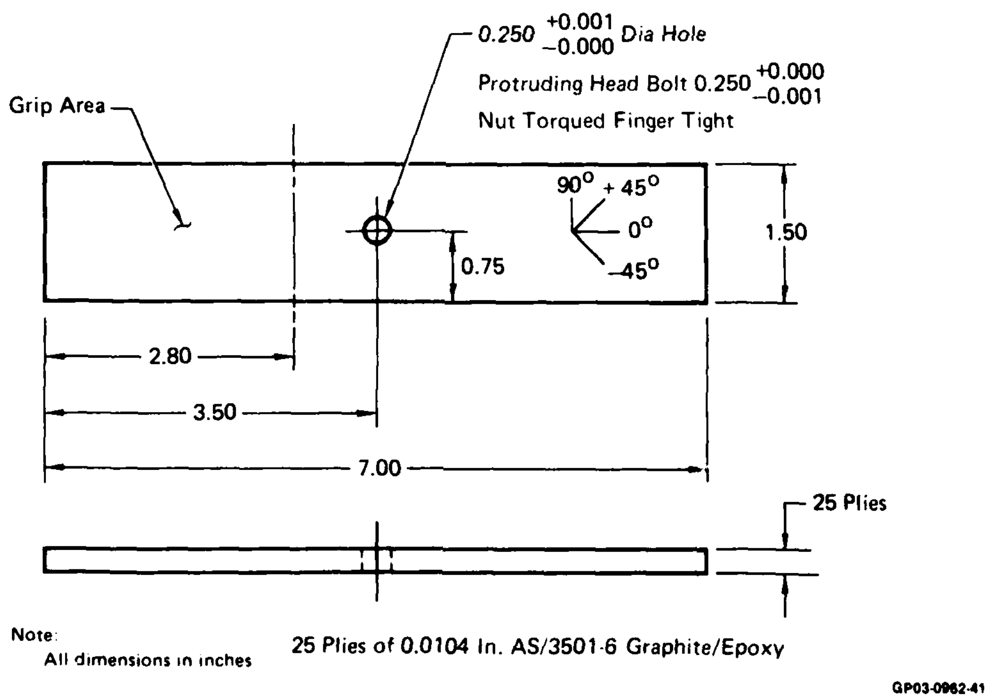


Figure 12. Test Specimen

A protruding head bolt was installed with close-tolerance into the fastener hole. The nut was torqued finger-tight.

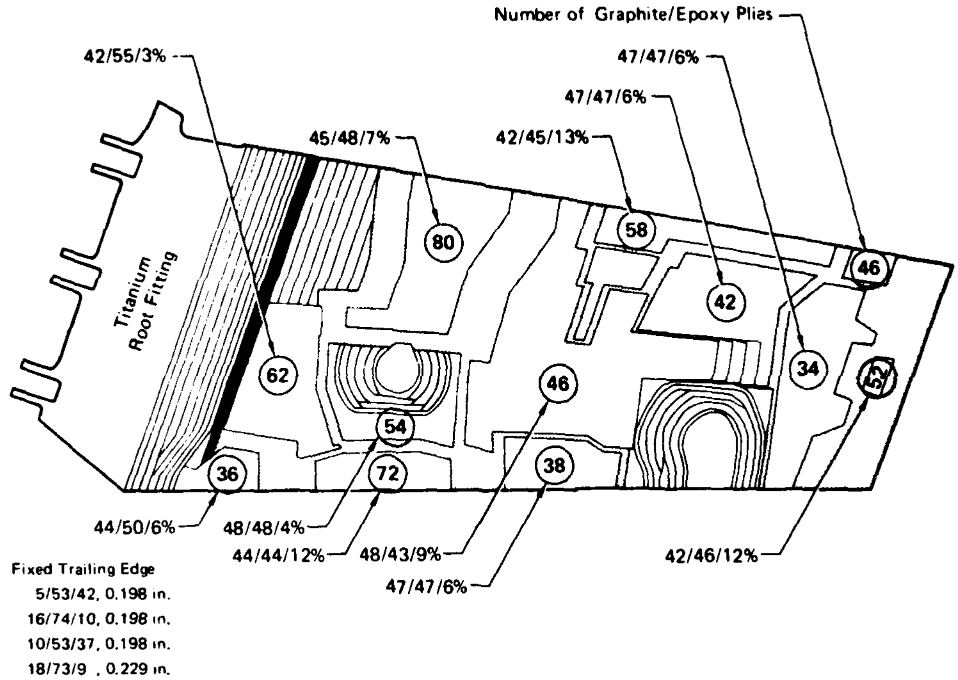
The grip length was determined as that necessary to carry the required load using the available serrated hydraulically actuated grips. The test section was sized such that anti-buckling guides were not necessary during testing in this compressive loading environment.

2. LAY-UP SELECTION - The structural design flexibility offered by composites leads to a multitude of ply lay-up arrangements in a structure. Examples are the lower and upper inboard wing skins and fixed trailing edge structure immediately aft of the rear spar of the F-18 aircraft (Figure 13). Fiber dominated lay-ups (laminates with high percentage of 0° plies) are used in wing torque box skin areas where the design is strength controlled. Matrix dominated lay-ups (laminates with a relatively high percentage of $\pm$ 45° plies) are used in stability critical structural components such as fixed trailing edge.

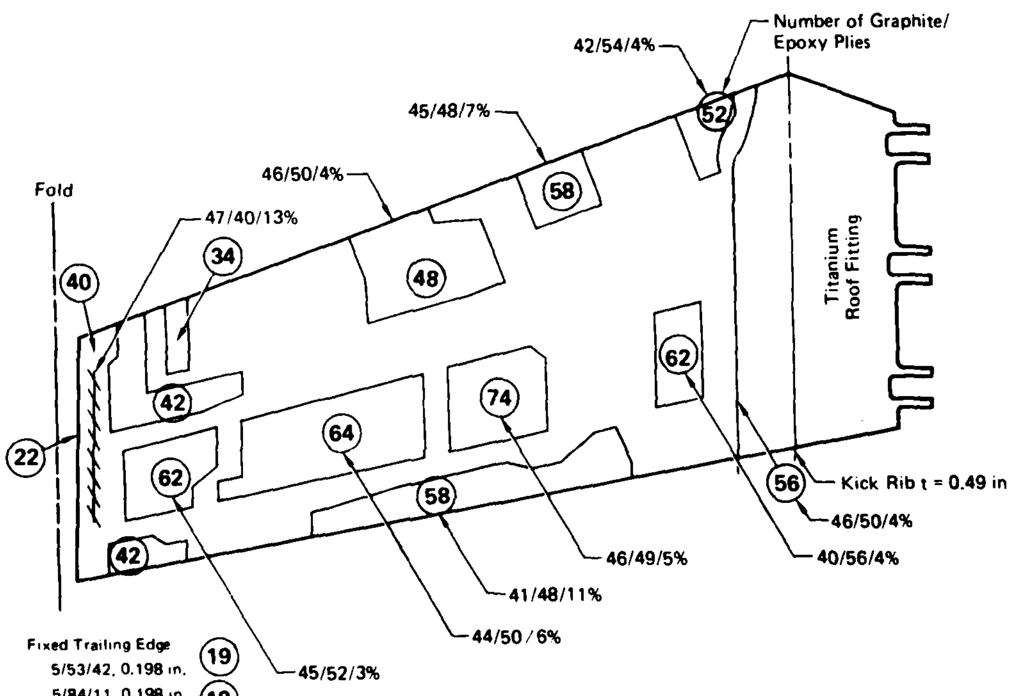
The baseline stress spectrum (Section II.2) is the F-15 Measured Mix spectrum for the upper skin of the inner wing. This is a compression dominated spectrum. The location used for computation of this baseline spectrum is the F-15 upper wing skin at the rear spar (Figure 14). A typical fiber dominated lay-up arrangement that would be used in the primary torque box in this area is 48/48/4 (48% 0° fibers, 48% $\pm$45° fibers, and 4% 90° fibers). A typical matrix dominated laminate which could be used for the fixed trailing edge area is 16/80/4.

At the rear spar, the wing torque box skin and the inboard trailing edge skin will have nearly the same span-wise strain spectra. Because of the significant difference in the elastic modulus of the two lay-ups that would be used in this area, the stress levels in the stress spectrum for the fiber dominated lay-up will be significantly higher than for the matrix dominated lay-up. For the purpose of minimizing variables in comparing experimental data, all specimens (both fiber and matrix dominated laminates) have the same thicknesses. A representative Gr/Ep composite material thickness in this area is approximately 0.25 inches. AS/3501-6 Gr/Ep was selected as the test material.

3. SPECIMEN MANUFACTURE - Four panels (two matrix dominated and two fiber dominated) were fabricated from AS/3501-6 graphite-epoxy prepreg of .0104 inches nominal thickness. The stacking sequence of these laminates are $[+45°, -45°, 0°, 0°]_3$, 90°, $[0°, 0°, -45°, +45°]_3$ (fiber dominated), and $[+45°, -45°, 0°, +45°, -45°]_2$, +45, -45°, 90°, -45°, +45°, $[-45°, +45°, 0°, -45°, +45P]_2$ (matrix dominated). The laminates were initially inspected ultrasonically (C-scan). These ultrasonic inspections indicated that the laminates were acceptable. Subsequently, the panels were machined into test specimens. Each specimen was inspected ultrasonically, discarding the ones showing defects caused by machining.



Lower Inboard Wing Skin



Upper Inboard Wing Skin

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Figure 13. F-18 Inboard Wing Skin Lay-Up Distribution

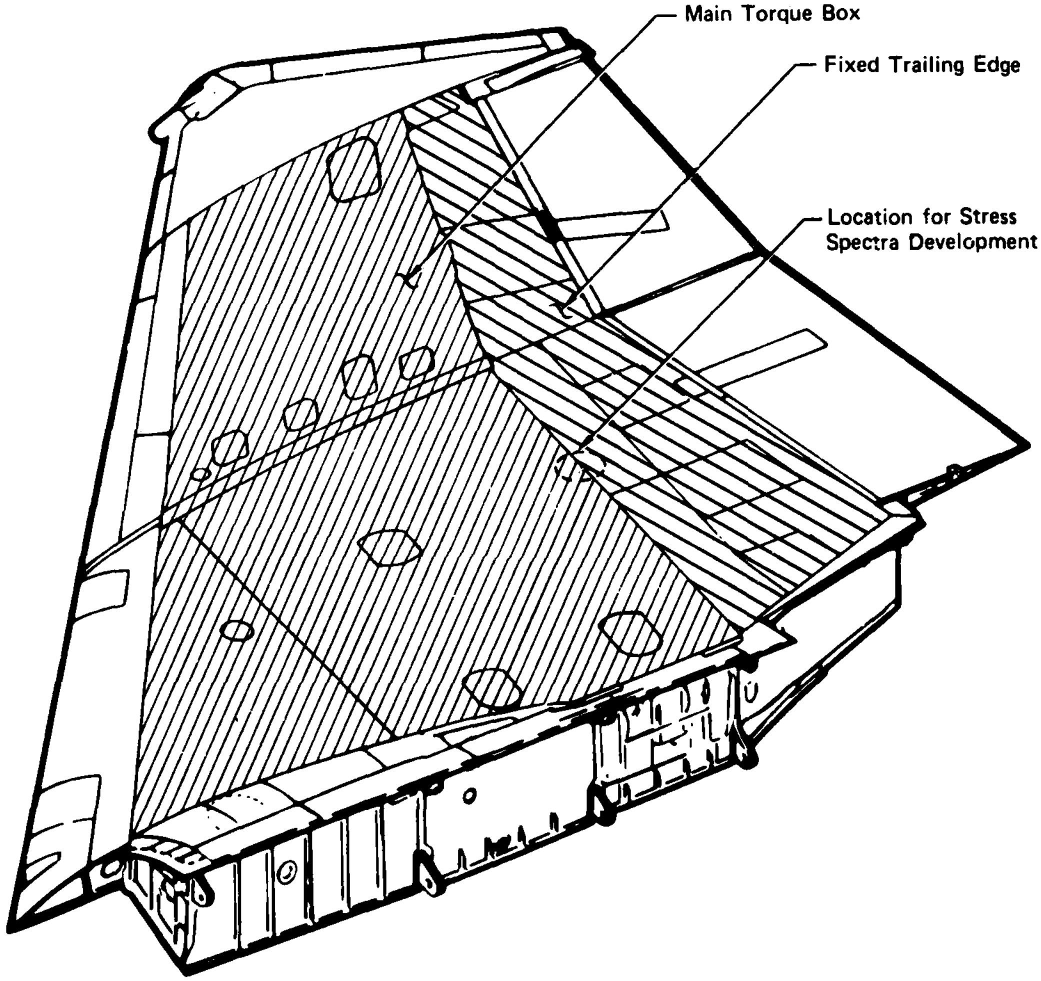

Figure 14. F-15 Wing Upper Skin Used as Study Base

SECTION IV

TESTING PROCEDURES

1. TEST PROGRAM SUMMARY - The program consisted of testing a total of 246 specimens divided over two phases (1) methodology test program and (2) spectrum variation study. The purpose of the methodology test program was to develop static strength, constant amplitude and baseline spectrum life information necessary to make life predictions for the spectrum variations phase. Details of the methodology test program are found in Section V. Details of the Spectrum Variation study are found in Section VI with data tabulations in Appendix B.

2. TEST TECHNIQUES

a. Test Procedures - All fatigue testing was performed in MTS Systems. Load application is controlled through mini-computers. Load feedback signals were monitored with strip chart recorders. In addition, dynamic comparisons of feedback measured loadings and programmed loadings were made to assure loading accuracies. Specimens were clamped in hydraulic grips. Buckling guides were not utilized. All fatigue loads were applied with test frequency controlled between 7 to 10 Hz.

All specimens were tested at room temperature in laboratory air, with as-manufactured moisture content.

b. Test Limit Stress Selection - Composite structures inherently have excellent fatigue properties. As a result, the stress levels used in the test program were significantly higher than would ordinarily be used in aircraft design. The test limit stress for the fiber dominated lay-up was chosen to be 85% of ultimate strength, and for the matrix dominated lay-up was chosen to be 66.7% of ultimate strength. In both cases, the estimated ultimate strength was the mean of a small sample of element test data.

In most applications, design values of ultimate strength would be established based on statistical analyses of a larger number of specimens with additional reductions in design values assumed, to account for uncertainties such as temperature, moisture level, and structural complexities. The resulting design values would be approximately 50% of the mean ultimate strength, rather than the values of 85% and 66.7% used in this program. Use of 50% would result in a test limit stress of approximately 42 ksi for the fiber dominated lay-up, and 27 ksi for the matrix dominated lay-up. These stress levels would have resulted in lives exceeding 1,000,000 spectrum hours. Test times would have been extremely long, and the lives would have been much greater than of interest in fighter aircraft design.

c. Failure Criteria - Since the tensile loads in the history were rather small, the failures were not of the type normally associated with specimen separation. The primary failure mechanism was matrix cracking at the fiber-matrix interface within a ply. This was followed by delamination in areas which have accumulated extensive matrix cracking. Interlaminar matrix cracking and delamination interacted to produce eventual crushing of the test section through the hole. Failure was defined as the point at which the specimen could not carry the compressive loads required by the test load history.

d. Determination of The Minimum Number of Replicate Specimens - A determination of the mean life in composite materials would normally require 20 or more duplicate specimens because of the normal scatter in fatigue life of composite materials. This number of specimens was not required for this program since the purpose was to delineate whether any abnormal changes to life were occurring with normal changes in usage. The argument was that abnormal changes in life would be readily apparent by changes in the mean life and that these changes would be determined through testing of five specimens at a minimum. Thus, a minimum of five specimens were tested at any condition to define whether the spectrum variation life was significantly different from the baseline behavior.

e. Moisture Conditions - It is known that moisture and temperature have adverse effect on the ultimate strength and constant amplitude fatigue life of composites. An analysis of the moisture absorption ability of AS/3501-6 graphite/epoxy indicates that very long exposure times at air-conditioned laboratory environments are necessary to reach high levels of moisture at the mid-plane of the specimens that could adversely influence the performance of composites. It was concluded that any casual moisture absorption variations due to waiting in a queue in controlled environment were not the primary source of strength variations during test.

f. Specimen Selection - Six composite panels were prepared for test in this program. Specimen lay-out is shown in Appendix D. Selection of specimens was controlled by a simple random process.

g. Quality Control - McDonnell-Douglas Quality Control specifications were followed during manufacture and machining. C-Scan and X-Ray photographs were made for each specimen both after panel preparation and after final machining. Discrepant specimens were discarded.

SECTION V

FATIGUE LIFE PREDICTIONS

The development of a spectrum fatigue life prediction methodology for graphite-epoxy composite structures was based on a coupling of testing and analysis development. The testing used to develop this prediction methodology preceded the testing described in Section VI.

1. METHODOLOGY DEVELOPMENT TEST PROGRAM - Constant amplitude and spectrum fatigue data were obtained for three different lay-ups: fiber dominated, matrix dominated, and intermediate. The geometry of the test specimen is shown in Figure 12.

A total of 88 specimens were tested. Eighteen specimens were statically tested to determine ultimate strength, 78 were tested in constant amplitude fatigue at different R-ratios, and 18 were subjected to spectrum fatigue loads.

Results of the ultimate strength tests are presented in Table 4. Results of the constant amplitude fatigue tests (Figure 15) indicate that the life of graphite-epoxy composites is dependent upon the lay-up and the R-ratio. In order to reduce or eliminate R-ratio testing for every lay-up that might be considered, a correlative methodology described in the following section was developed.

TABLE 4. METHODOLOGY DEVELOPMENT TEST PROGRAM
ULTIMATE STRENGTH RESULTS

Lay-Up	48/48/4			25/67/8			16/80/4		
	Load (lb)	Strain (μ · in./in.)	Stress (ksi)	Load (lb)	Strain (μ · in./in.)	Stress (ksi)	Load (lb)	Strain (μ · in./in.)	Stress (ksi)
Compression	31,200	8,520	80.00	24,200	9,550	64.64	20,700	11,190	53.08
	31,600	7,770	81.03	24,200	9,565	64.64	18,400	9,440	47.18
	29,600	7,740	75.89	22,800	9,080	60.90	20,100	10,665	51.54
Average	30,800	8,010	79.00	23,733	9,398	63.39	19,730	10,432	50.60
Tension	25,400	5,835	65.13	14,600	5,435	39.00	14,800	7,350	37.95
	24,600	5,735	63.08	16,800	6,100	44.87	14,700	7,175	37.69
	22,400	5,345	57.44	16,800	6,010	44.87	15,000	7,355	38.46
Average	24,130	5,638	61.87	16,066	5,848	42.91	14,830	7,293	38.08

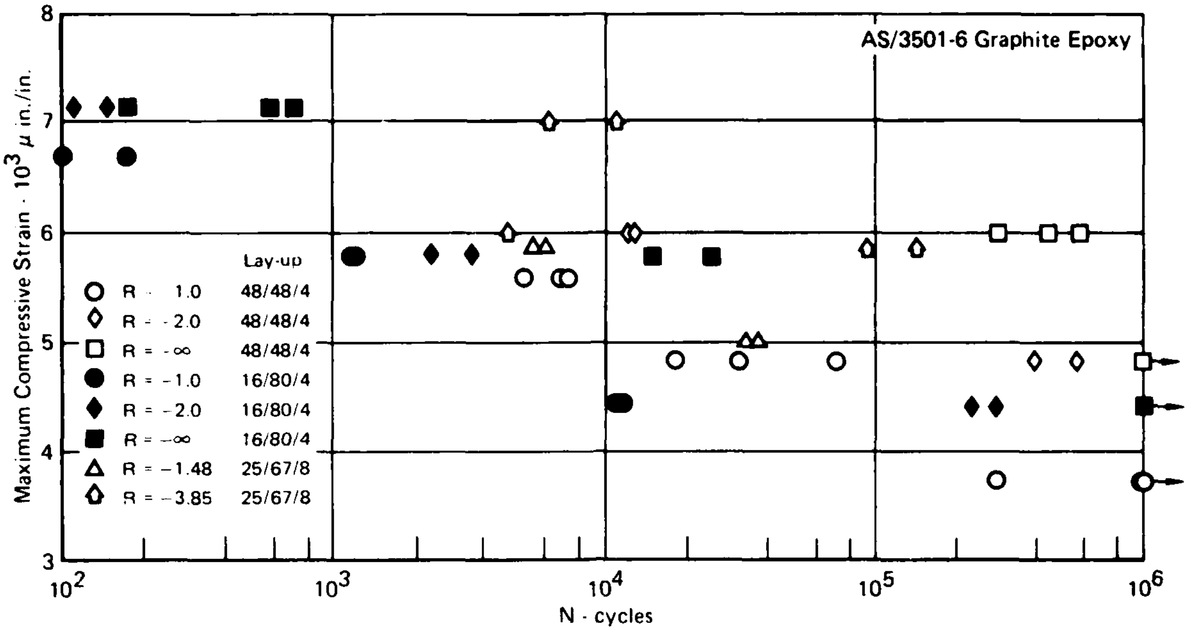

Figure 15. Methodology Development Test Program
Constant Amplitude Results

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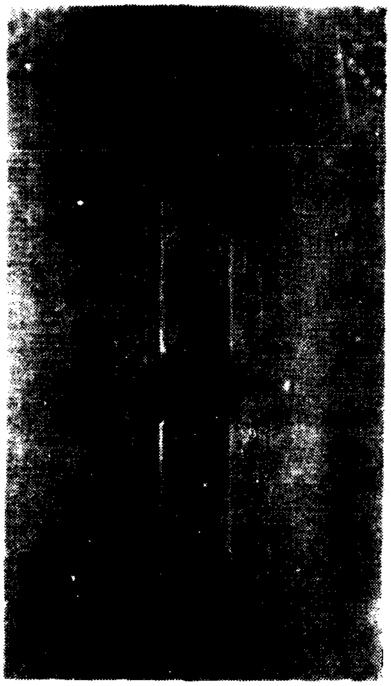
A number of specimens were examined nondestructively by x-ray photography to observe the type and location of damage during different stages of their fatigue life. Since the regular x-ray image of damaged specimens do not provide discernable contrast for examination, the test specimens were coated by a mixture of tetra-bromoethane (TBE) and die penetrant (ZL-2A). The fluid mixture consists of 25% TBE and 75% die penetrant. The fluid was applied only at the edges of the specimens for the purpose of detecting edge delaminations. After completion of the first step, the fluid was also applied at the fastener hole.

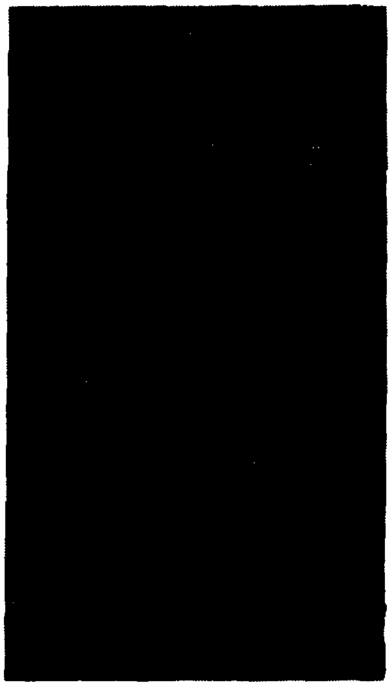
Figure 16 contains x-ray photographs of the fiber dominated (48/48/4) specimen with only the edge coated with the TBE & ZL-2A fluid. In Figure 16-A, the maxtrix cracking in the 90° ply can be seen as fine horizontal lines. The penetrant appears to have followed the 90° ply cracks to the matrix cracks in the adjacent 0° ply. These 0° ply cracks can be seen as fine vertical lines at the side of the fastener hole. There is no indication of edge delamination. Figure 16-B shows the matrix cracks in the 90° ply, at the centerline of the specimen.

Figure 17 contains x-ray photographs of the same specimen after the fastener hole was coated with the penetrant fluid. Figure 17-A shows in detail the accumulated damage around the periphery of the hole. The distribution of matrix cracking thru the thickness can be seen in Figure 17-B.



Specimen P133 1.5 X

16-A



Specimen P133 1.5 X

16-B

$R = -1,\ 0.65\ F_{cu}$ 13,000 Cycles (Half Life)

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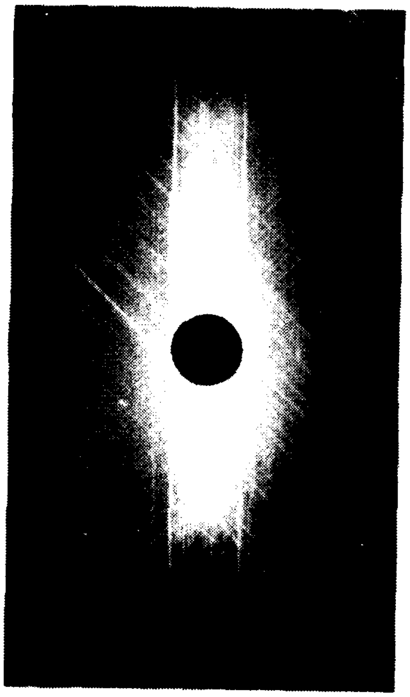
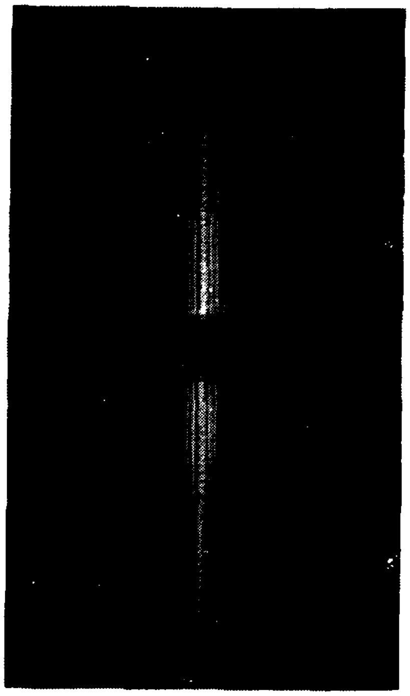
Figure 16. X-Ray Photographs of Fiber Dominated Lay-Up with Penetrant Applied Only to Specimen Edges



Specimen P133 1.5 X

17-A



Specimen P133 1.5 X

17-B

13,000 Cycles (Half Life)

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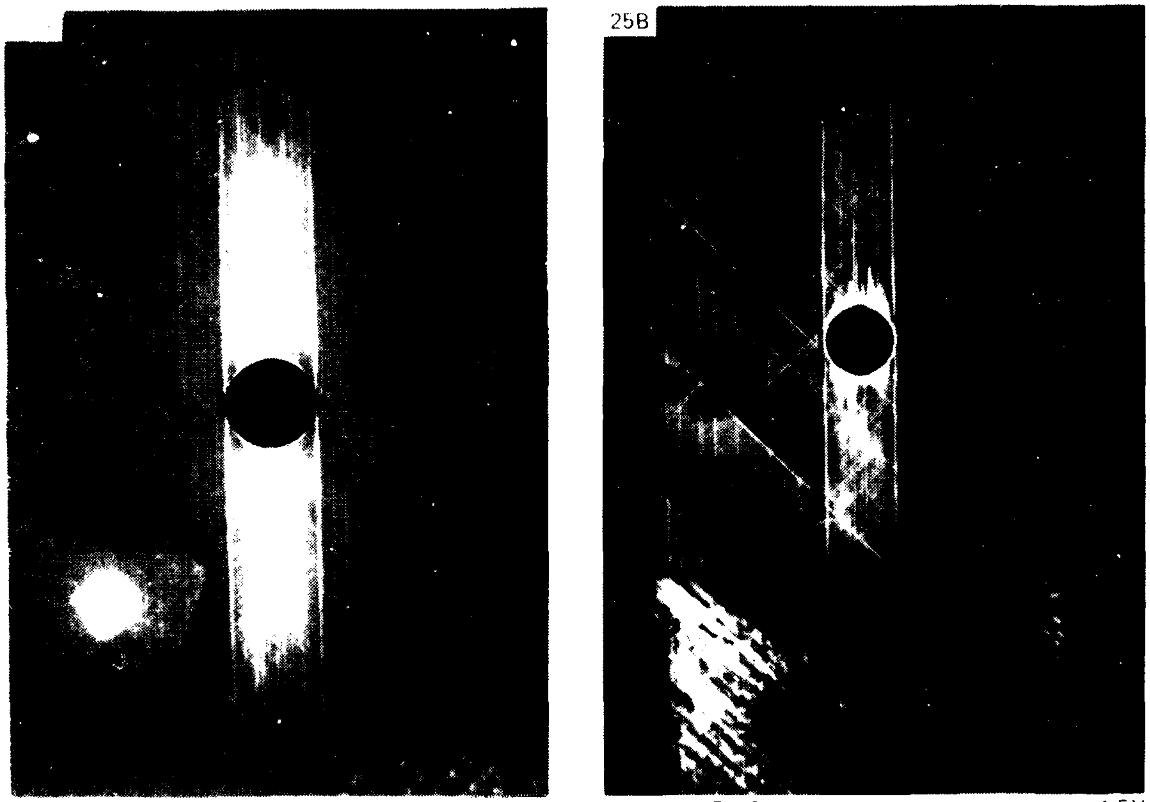
Figure 17. X-Ray Photographs of Fiber Dominated Lay-Up with Penetrant Applied to Specimen Edges and Holes

Figure 18 contains x-ray photographs of another fiber dominated speciman where both the edges and hole were coated with the TBE and ZL-2A fluid. The figure demonstrates the progression of matrix cracking during the different stages of fatigue life.

Figure 19 contains x-ray photographs of a matrix dominated (16/80/4) specimen taken with only the edges of the specimen coated with the fluid. In Figure 19-A, the matrix cracking in the 90° ply can be seen as fine horizontal lines. It demonstrates absence of edge delamination. Figure 19-B shows the matrix cracks in the 90° ply, at the centerline of the specimen. Figure 20-A is an x-ray photograph of this same specimen with the hole also coated with the x-ray opaque fluid. Figure 20-B is an x-ray photograph of another specimen, similiarly tested. The similiarity of the two photographs indicates the pattern of matrix cracking is a repeatable process. These figures indicate that the damage is concentrated at the hole.



Specimen P119 | Specimen P119 | 1.5X

18-A | 18-B

75,000 Cycles (Quarter Life) R = ∞, 0.8 F_{cu} 150,000 Cycles (Half Life)

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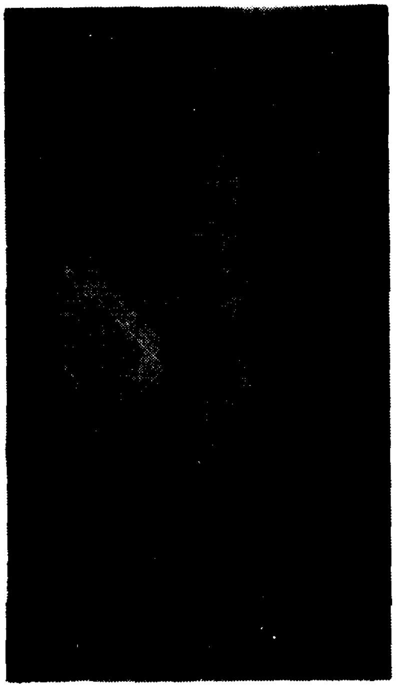
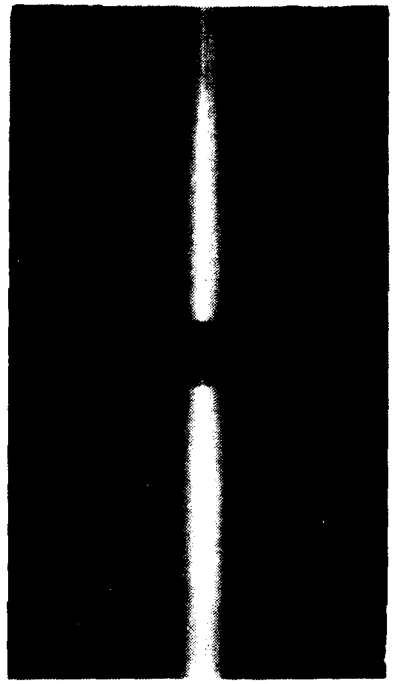
Figure 18. X-Ray Photographs of Fiber Dominated Lay-Up at Quarter and Half Life with Penetrant Applied to Specimen Edges and Hole



Specimen P221 1.5 X

19-A



Specimen P221 1.5 X

19-B

R = −1, 0.50 F_{cu} 5,000 Cycles (Half Life)

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Figure 19. X-Ray Photographs of Matrix Dominated Lay-Up with Penetrant Applied Only to Specimen Edges

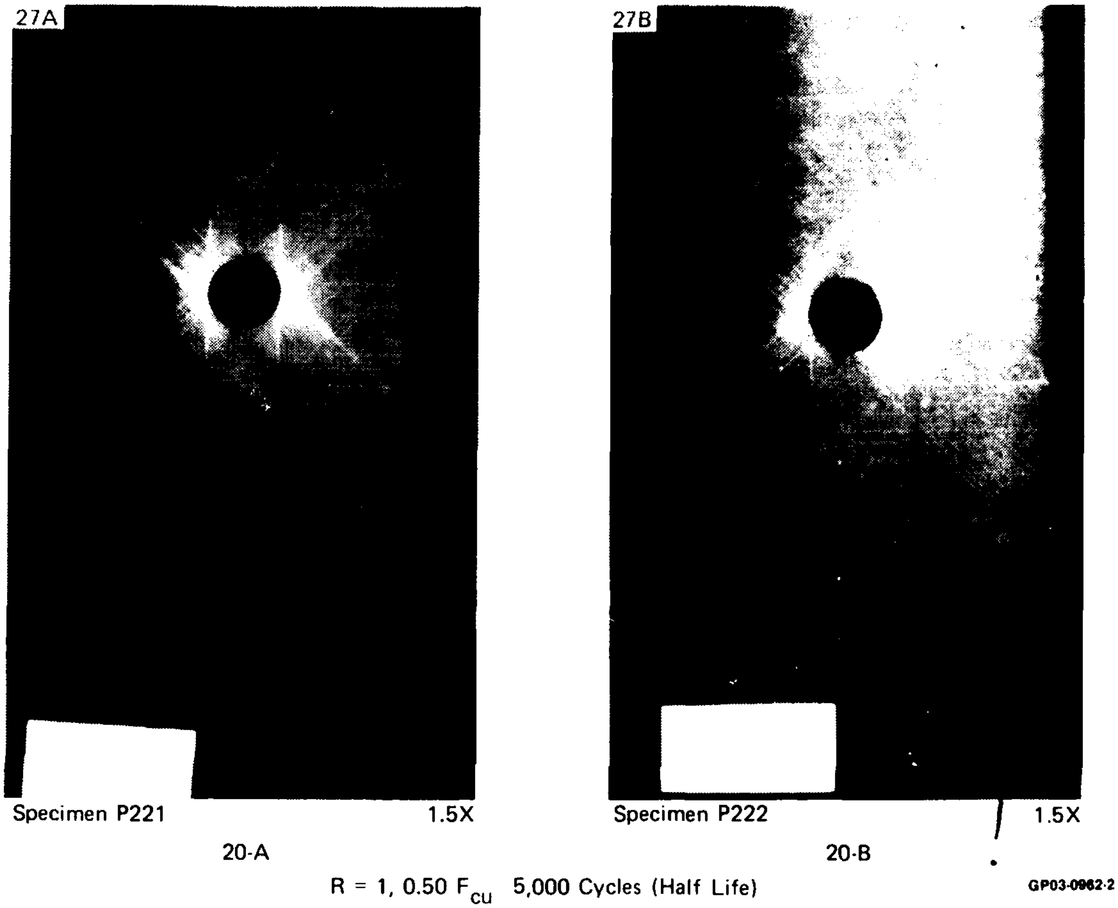


Figure 20. X-Ray Photographs of Matrix Dominated Lay-Up for Two Specimens with Penetrant Applied to Specimens Edges and Holes

2. LIFE PREDICTION METHODOLOGY

The data presented in Figure 15 indicate that to predict the fatigue life of composite structures it is essential to account for lay-up variations and load ratios. A correlation parameter was developed based on the concept of strain energy density factor (References 4 and 5). The strain energy stored in the volume element (dV = dxdydz) of an isotropic material is:

$$\frac{dU}{dV} = \frac{1}{2E}\left(\sigma_x^2 + \sigma_y^2 + \sigma_z^2\right) - \frac{\nu}{E}\left(\sigma_x \sigma_y + \sigma_y \sigma_z + \sigma_z \sigma_x\right) + \frac{1}{2\mu}\left(\tau_{xy}^2 + \tau_{xz}^2 + \tau_{yz}^2\right)$$

Substitution of stresses in terms of stress intensity factors results in the following simple relationship for strain energy density

$$\frac{dU}{dV} = \frac{S}{r}$$

where r is the distance from a crack tip, and S is a quadratic function in the form

$$S = a_{11}k_1^2 + 2a_{12}k_1k_2 + a_{22}k_2^2 + a_{33}k_3^2$$

and is a measure of the intensity of the strain energy density field around the crack tip. The coefficients a_{ij} are

$$a_{11} = \frac{1}{16\mu}\left[(1 + \cos\theta)\ (\kappa - \cos\theta)\right]\frac{1}{\cos\omega}$$

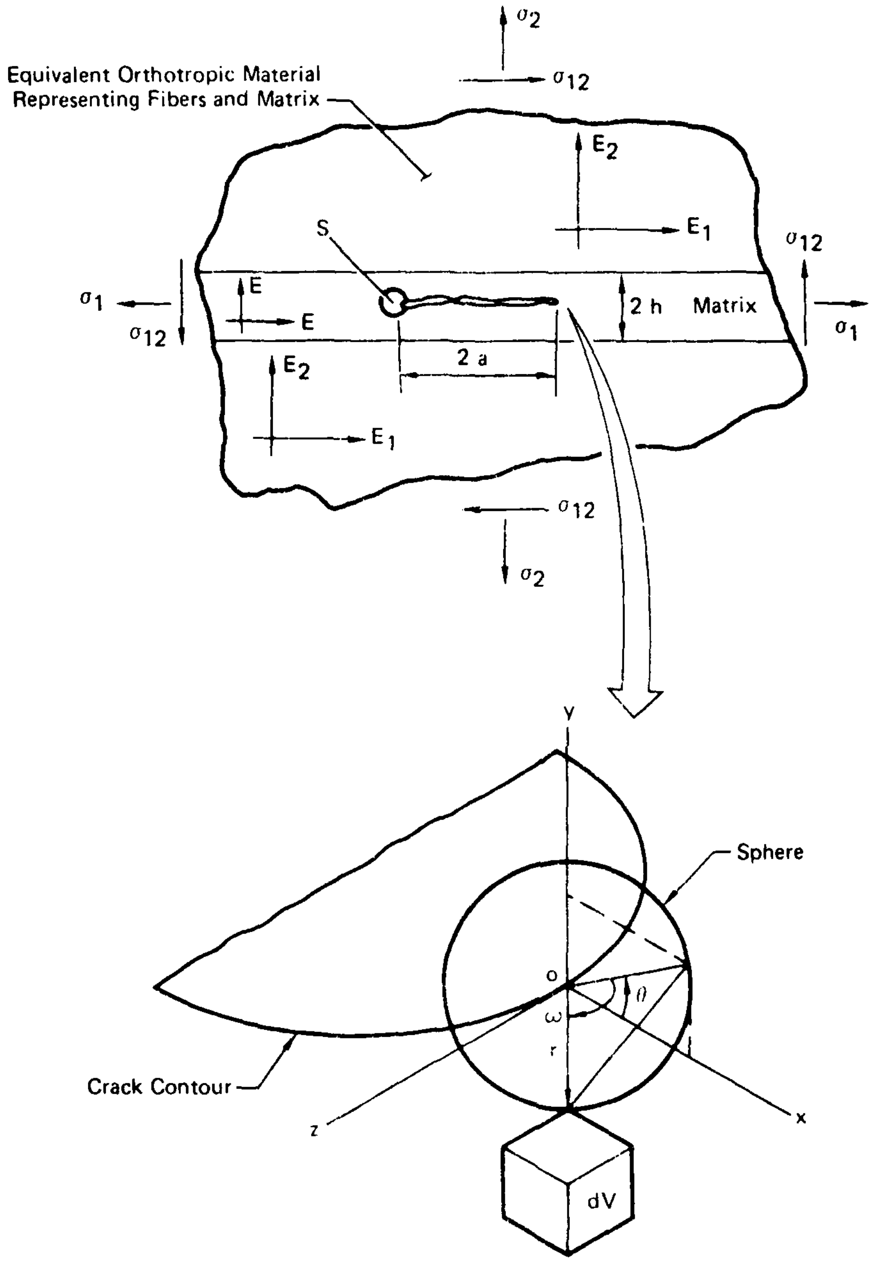
$$a_{12} = \frac{1}{16\mu}\left[\sin\theta\ (2\cos\theta - (\kappa - 1))\right]\frac{1}{\cos\omega}$$

$$a_{22} = \frac{1}{16\mu}\left[(\kappa + 1)(1 - \cos\theta) + (1 + \cos\theta)(3\cos\theta - 1)\right]\frac{1}{\cos\omega}$$

$$a_{33} = \frac{1}{4\mu}\ \frac{1}{\cos\omega}$$

The elastic constant κ is equal to $(3-4\nu)$ for plane strain and $(3-\nu)/(1+\nu)$ for plane stress (μ and ν are the shear modulus and Poissons ratio respectively). The angles and describe the crack trajectory and are defined in a special spherical coordinate system shown in Figure 21.

An analytic model was developed with the assumption that the matrix is the weak link of the composite systems. This is modeled as an isotropic layer of resin containing a through-the-thickness crack sandwiched between the edges of two semi-infinite orthotropic plates as shown in Figure 21. E and ν are the elastic constants of the resin, while E_1, E_2, ν_{12}, and μ_{12} are the elastic constants of the equivalent orthotropic material surrounding the resin strip of width 2h. The subscripts 1 and 2 represent directions parallel and perpendicular to the ply fibers, respectively. The strip width 2h is directly related to the fiber volume fraction. For low fiber volume fraction this width can be estimated to be $h = R(\frac{1}{2}\sqrt{\frac{\pi}{V_f}} - 1)$, where R is the fiber radius and V_f is the fiber volume fraction. However, for a graphite-epoxy composite system such as AS/3501-6 with high fiber volume fraction, the strip width 2h is very small, hence the limiting value $(h/a \rightarrow 0)$ can be used.



A Special Spherical Coordinate System

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Figure 21. Cracked Lamina Model

The intensity of the crack tip stress field for this problem can be written as

$$k_1 = \lambda_1 \sigma_2 \sqrt{a}$$

$$k_2 = \lambda_2 \sigma_{12} \sqrt{a}$$

where σ_2 and σ_{12} are the far field normal and tangential stresses, a is half the crack length, and λ_1 and λ_2 are correction functions which depend on fiber spacing and elastic constants. Values of λ_1 and λ_2 for AS/3501-6 graphite epoxy are

$$\lambda_1 = .2843$$
$$\lambda_2 = .0911$$

Far field stresses σ_2 and σ_{12} are obtained through lamination plate theory which assumes uniform strain distribution through the thickness of a laminate. This assumption reduces the three dimensional problem to a two dimensional one, which simplifies the formulation of the strain energy density factor S.

In the development of this preliminary model, for simplicity, it was assumed that the through-the-thickness crack grows parallel to adjacent fibers. This assumption permits omission of the interaction term $2a_{12}\ K_1\ K_2$ from the strain energy density factor equation. The relationship for the simplified strain energy density factor is

$$S = a_{11}\ k_1^2 + a_{22}\ k_2^2$$

Strain energy density factors for each ply are summed as a damage indicator parameter, and normalized with respect to ultimate compressive or tensile strength of the laminate to obtain the following relationship

$$\bar{S} = \left(\frac{f_i}{F_{u_i}}\right)^2 \sum_{n=1}^{N} (S_{\text{per ply}})_i + \delta_{ij} \left(\frac{f_j}{F_{u_j}}\right)^2 \sum_{n=1}^{N} (S_{\text{per ply}})_j$$

i, j = compression, tension

$$\delta_{ij} = \begin{cases} +1 & i \neq j \\ -1 & i = j \end{cases}$$

The parameter $\bar{S}$ was then used to correlate constant amplitude fatigue data for the three laminates. Results, shown in Figure 22, indicate correlation of fatigue life for different lay-ups for a given R (f_{min}/f_{max}) ratio and value of $\bar{S}$. Subsequently, an empirical relationship (Figure 23) was developed to account for R-ratio effect. This empirical relationship, when applied to the data presented in Figure 22, resulted in a correlation of $\bar{S}$ and life that was independent of R ratio, as demonstrated in Figure 24.

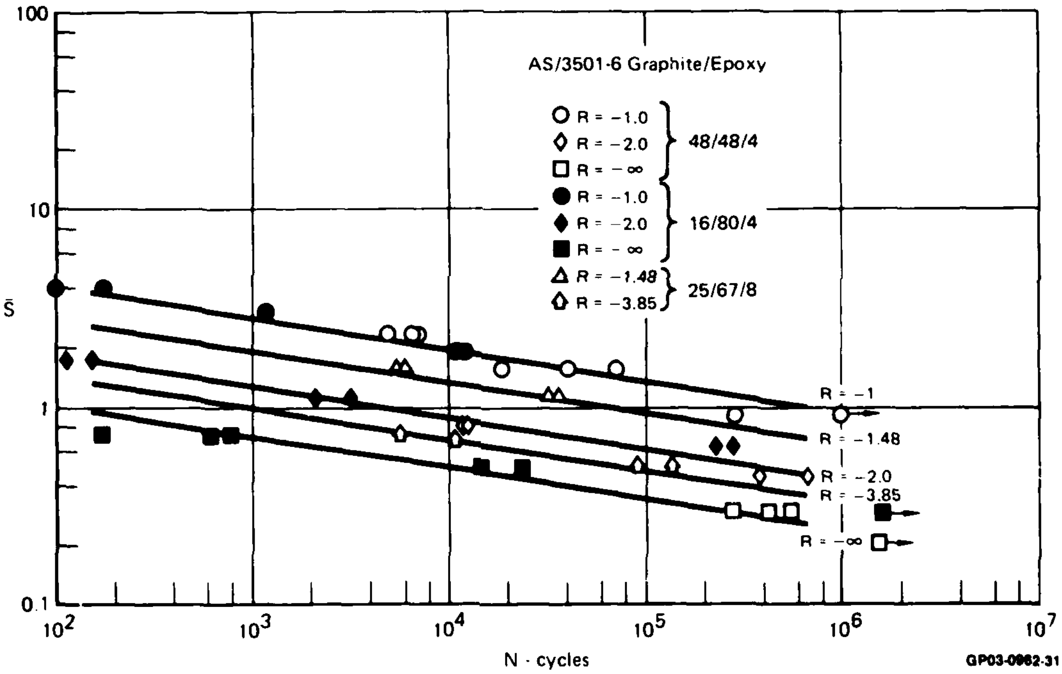

Figure 22. Correlation of Strain Energy Density with Fatigue Life

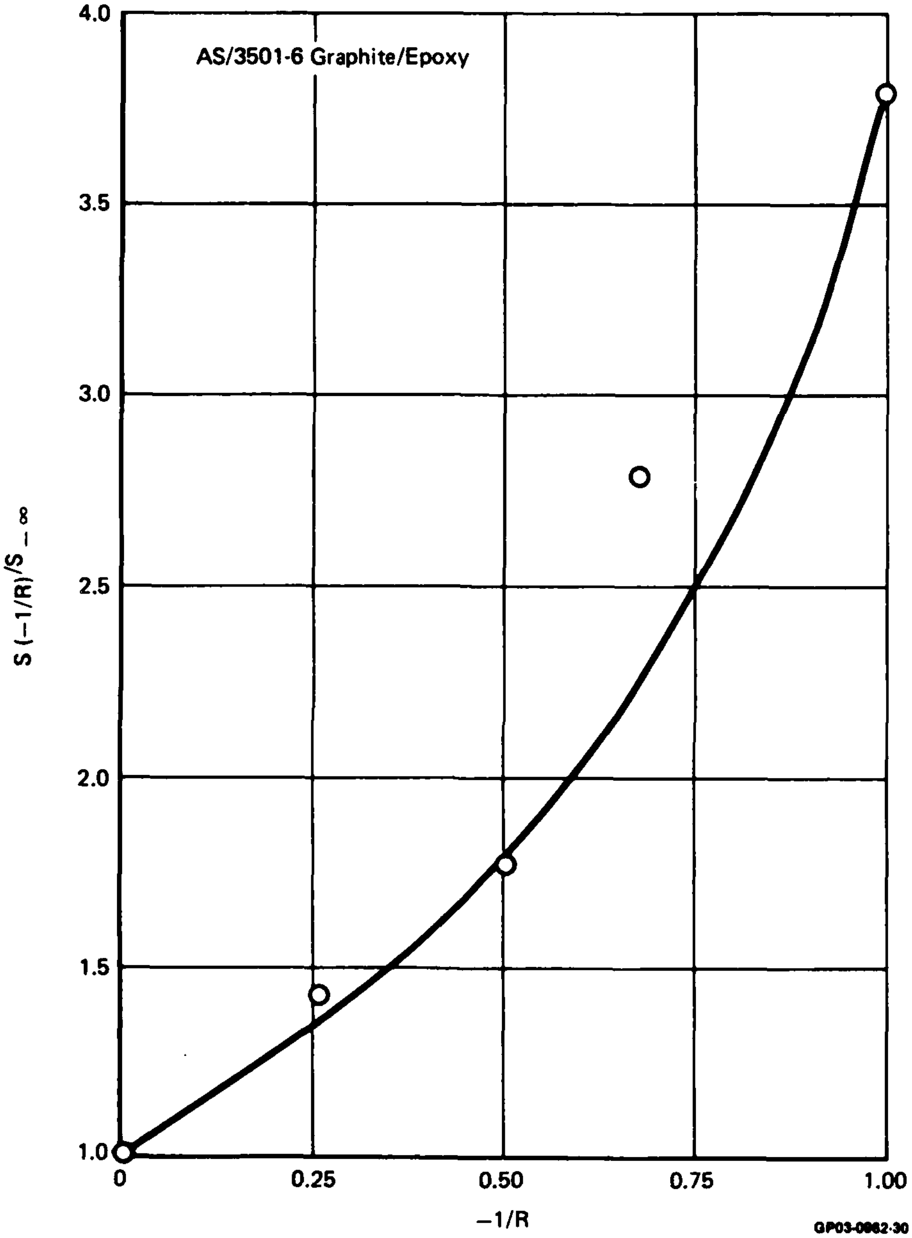

Figure 23. Stress-Ratio Correction

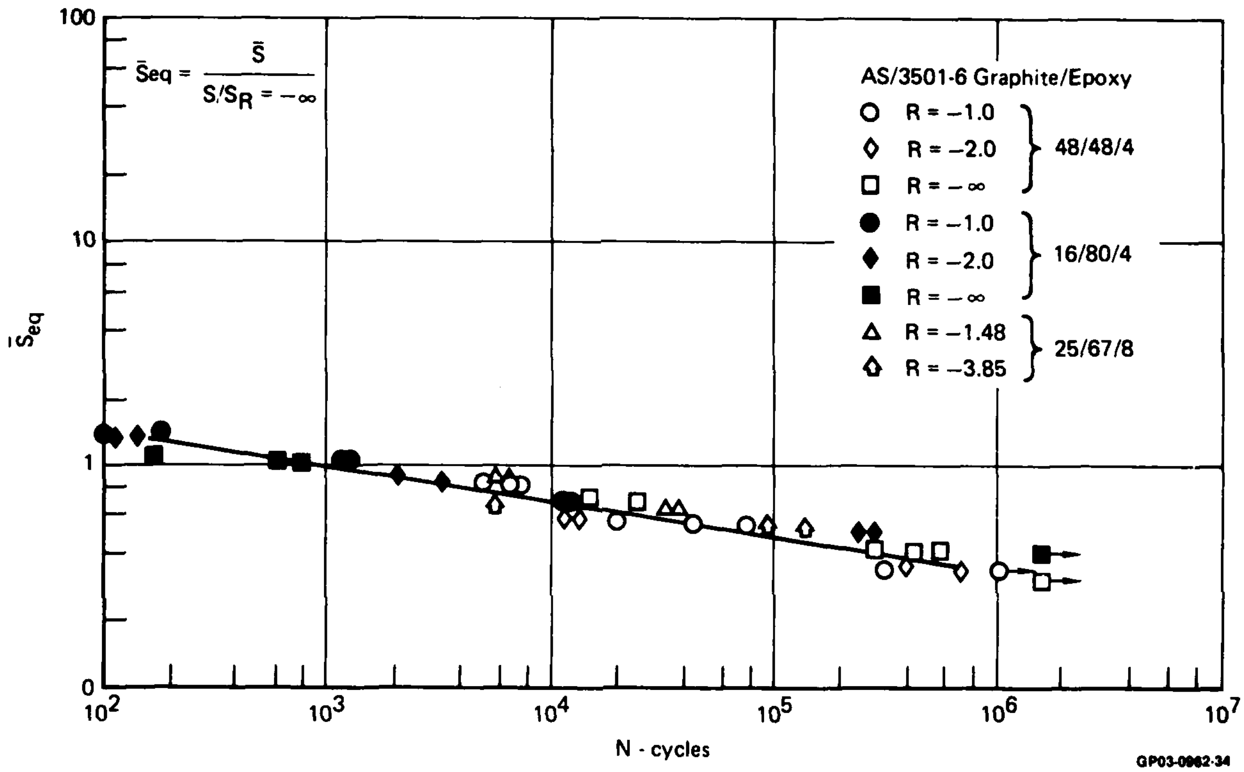

Figure 24. Correlation of Equivalent Strain Energy Density with Fatigue Life.

This constant amplitude predictive methodology was then used in conjunction with two linear fatigue damage models to predict spectrum fatigue life. The fatigue damage models used are linear strength reduction (Reference 6) and Miner's Rule. The linear strength reduction model (Figure 25) is a modified version of Miner's Rule with the assumption of a linear reduction in the residual compressive strength due to cyclic loading.

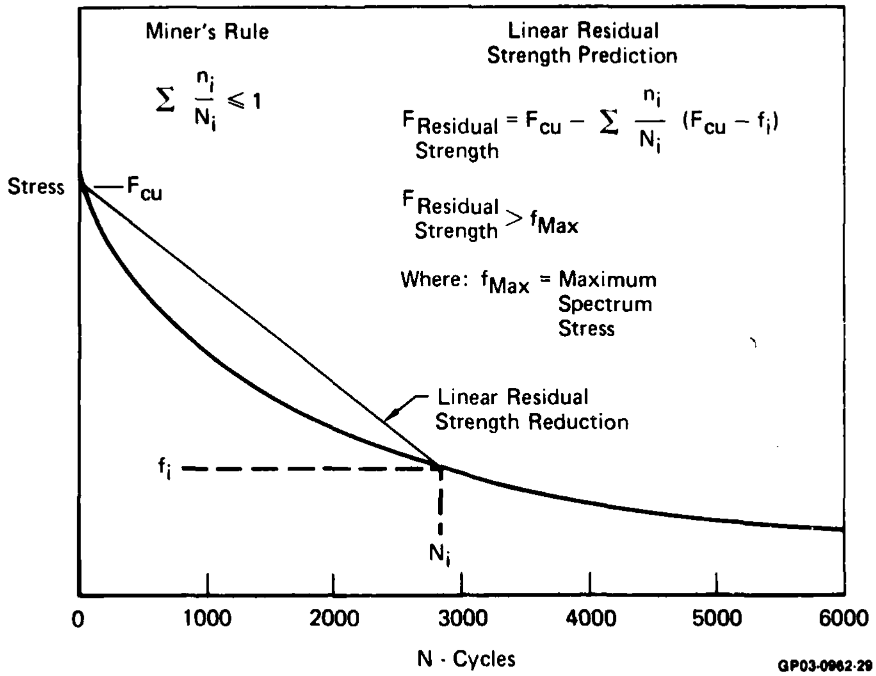

Figure 25. Linear Fatigue Damage Models

The linear strength reduction model in conjunction with the normalized strain energy density factor was used to predict fatigue lives of three different lay-ups subjected to the spectrum loads represented in Figure 26. These spectra are derivatives of the F-15 design mix baseline stress spectrum. The F-15 design mix spectrum was truncated to 55 percent of Test Limit Stress (TLS) to develop the truncated F-15 design mix. The same spectrum was modified by increasing the tension (negative) load values to obtain the increased tension spectrum. The increased tension spectrum has a maximum tension load approximately 2/3 of the maximum compression load. Correlations of test results and predictions are shown in Figure 27 along with the experimental data. The comparison between linear strength reduction model and Miner's Rule for the matrix dominated lay-up subjected to increased tension spectrum is shown in Figure 28. This comparison shows Miner's Rule to be substantially unconservative for higher values of test limit stress. All subsequent analyses were performed using the linear strength reduction model.

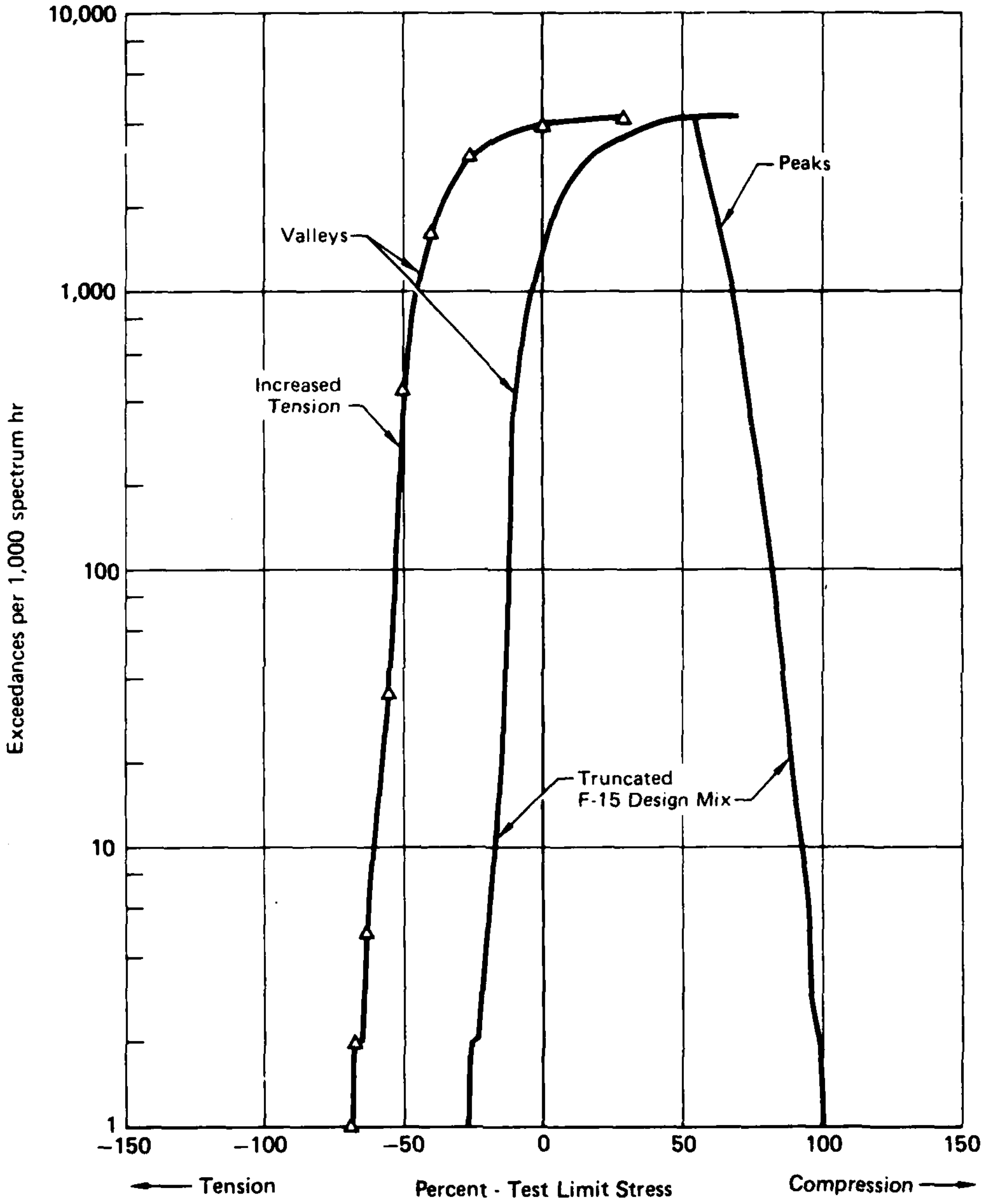

Figure 26. Methodology Development Test Spectra

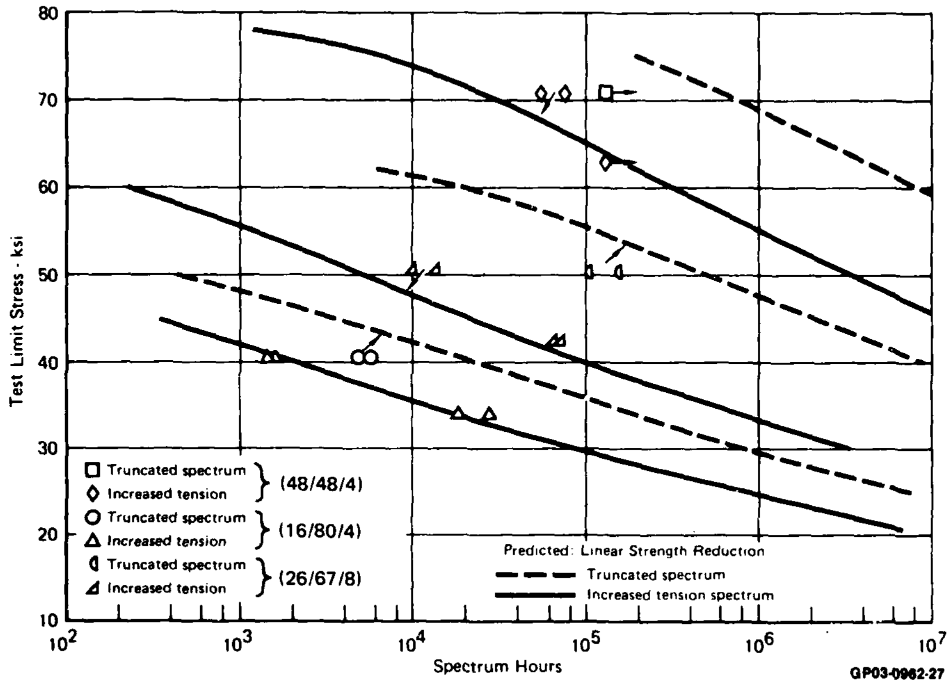

Figure 27. Methodology Development Test Program - Results of Spectrum Fatigue Tests

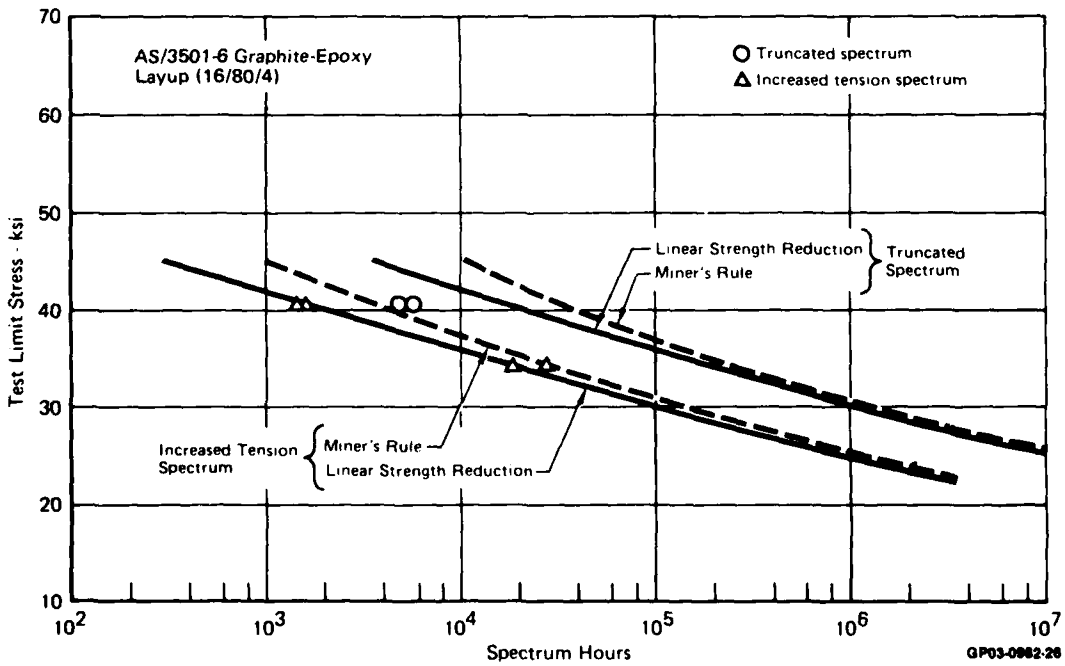

Figure 28. Comparison of Linear Strength Reduction and Miner's Rule Predictions

SECTION VI

FATIGUE LIFE CORRELATIONS

1. SUMMARY - The spectra variation types considered were: (a) clipping to 90% test limit stress, (b) addition of either 115% or 125% test limit stress overloads, (c) addition of low loads, (d) truncation to 70% test limit stress, (e) clipping of tension loads, (f) air-to-ground baseline, (g) increased severity and number of air-to-air loads, and (h) variations in test limit stress.

Comparisons of predicted and measured life are presented in Tables 5 and 6. Also presented in those tables are comparisons with baseline life (F-15 Measured Mix-Truncated), and Weibull statistical parameters. In general, predictions and test results are in agreement within a factor of three. Generally, predicted life was greater than test life. The overall average ofredicted/ test median life was 2.08 for the fiber dominated lay-up, and 1.46 for the matrix dominated lay-up. Figure 29 summarizes comparisons of test and predicted life, including confidence limits on test lives based on Weibull statistical analyses. Even with the uncertainty in test mean considered, as shown by the range between 90% confidence limits, predicted lives are generally greater than test lives.

TABLE 5. TEST SUMMARY

Fiber Dominated Lay-Up (48/48/4)

Spectrum Variation	Predicted Life	Test Life (Mean)	Predicted Life / Test Life	Predicted Life / Predicted Baseline Life	Test Life / Test Baseline Life	Number of Specimens	Weibull Slope β	Weibull Theta Life	Standard Deviation Log Life
F-15 Measured Mix Truncated (Baseline)	289,000	198,000	1.46	1.00	1.00	5	6.84	209,000	0.067
Clipping to 90% Test Limit Stress	540,000	227,000	2.38	1.87	1.15	5	6.91	240,000	0.069
Addition of 115% Test Limit Stress Overloads	7,440	4,180	1.78	0.03	0.02	8	0.95	6,150	0.564
Addition of Low Loads to Match Original Measured Mix	286,000	102,000	2.80	0.99	0.52	5	2.13	122,000	0.216
Truncation to 70% Test Limit Stress	362,000	138,000	2.62	1.25	0.70	6	0.81	217,000	0.597
Clipping of Tension Loads	263,000	176,000	1.49	0.91	0.89	5	2.20	207,000	0.208
90% of F_{cu}	92,200	52,900	1.74	0.32	0.27	12	1.20	71,700	0.430
80% of F_{cu}	775,000	328,000	2.36	2.68	1.66	5	4.31	357,000	0.111
Increased Severity and Number of Air to Air Loads	1,870	1,990	0.94	0.01	0.01	10	0.66	3,470	0.745
Air to Ground Baseline	2,130,000	667,000	3.19	7.37	3.37	6	1.96	804,000	0.248

Weighted Average 0.455

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TABLE 6. TEST SUMMARY

Matrix Dominated Lay-Up (16/80/4)

Spectrum Variation	Predicted Life	Test Life (Mean)	Predicted Life / Test Life	Predicted Life / Predicted Baseline Life	Test Life / Test Baseline Life	Number of Specimens	Weibull Slope β	Weibull Theta Life	Standard Deviation Log Life
F 15 Measured Mix-Truncated (Baseline)	51,500	130,000	0.40	1.00	1.00	15	1.00	188,000	0.512
Clipping to 90% Test Limit Stress	68,100	145,000	0.47	1.32	1.11	10	0.81	228,000	0.621
Addition of 125% Test Limit Stress Overloads	25,200	5,410	4.66	0.49	0.04	13	1.12	7,500	0.459
Addition of Low Loads to Match Original Measured Mix	51,100	78,800	0.65	0.99	0.60	27	1.02	113,000	0.526
Truncation to 70% Test Limit Stress	61,200	75,300	0.81	1.19	0.58	5	0.56	144,000	0.859
Clipping of Tension Loads	44,500	22,200	2.00	0.86	0.17	7	1.18	30,300	0.404
85% of F_{cu}	2,600	838	3.07	0.05	0.01	8	1.65	1,050	0.302
55% of F_{cu}	823,000	1,690,000	0.49	15.98	13.00	8	1.74	2,080,000	0.275
Increased Severity and Number of Air-to-Air Loads	8,500	4,810	1.76	0.16	0.04	12	1.52	6,120	0.384
Air-to-Ground Baseline	259,000	776,000	0.33	5.02	5.97	5	1.33	1,020,000	0.342

Weighted Average - 0.490

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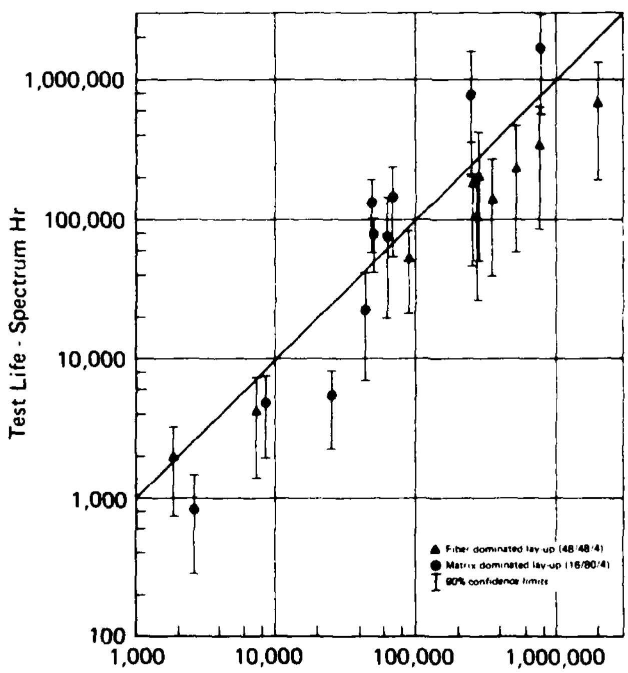

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Figure 29. Comparison of Predicted and Test Lives

In many design or analysis situations, the baseline test and predicted life are known, and the effect of a spectrum variation needs to be estimated through prediction. Figures 30 and 31 summarize the ability of the predictive methodology to estimate the effects of spectrum variations. As with the predictions of absolute life, the predictions of relative life are generally unconservative. The greatest inaccuracies occurred with the matrix dominated lay-up and with those spectrum variations causing reductions in life from the baseline spectrum life.

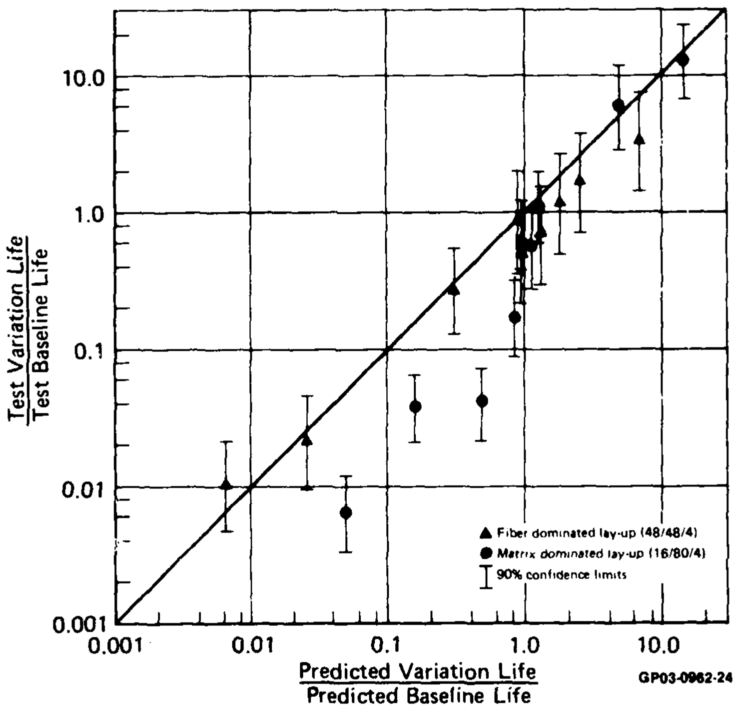

Figure 30. Comparison of Predicted and Measured Effect of Spectrum Variation on Life

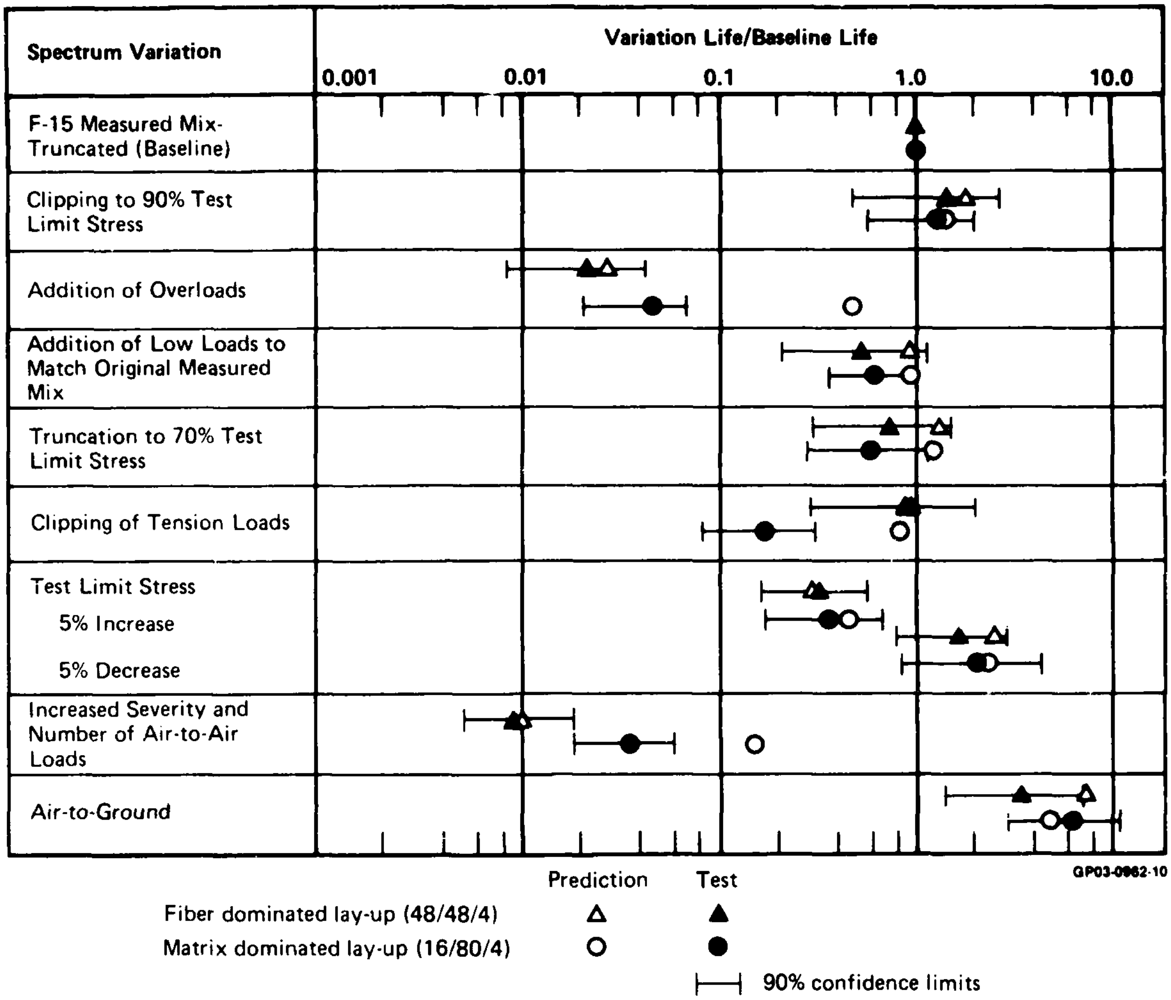

Figure 31. Effects of Spectrum Variation on Life

The data summarized in Figure 31 indicate the effects of spectrum variations on fatigue life is generally the same for both the fiber and matrix dominated lay-ups. Clipping to 90% Test Limit Stress was found in test to increase life, and this effect was predicted with reasonable accuracy. Addition of overloads was found in test to decrease life, and this effect was accurately predicted for the fiber dominated lay-up. The reduction in test life was similar for the matrix dominated lay-up, but the predicted life was considerably greater than test life. Improvements in prediction methodology appear necessary in this area.

Addition of low loads was found to reduce life, by a greater amount than predicted. Further testing would be required to determine whether this discrepancy is caused by test anomalies, or by prediction methodology error. Truncation to 70% test limit stress was found to reduce life. A modest increase in life was predicted, and this discrepancy of test and prediction is probably caused by test scatter. Further testing would be required to substantiate this conjecture.

Clipping of tension loads was found to decrease life, by a substantial amount for the matrix dominated lay-up. The predicted lives with this tension clipping were less than the baseline lives, whereas the predicted effect should be to increase life. Improvement in predictive methodology in this area is required. The substantial decrease in test life for the matrix dominated lay-up is probably caused by test scatter; further testing would be required to substantiate this conclusion.

The effects of changes in test limit stress were found in test to correlate well with predictions, for both the fiber and matrix dominated lay-ups.

Increasing the severity and number of air-to-air loads was found to decrease life, and the decrease was predicted for the fiber dominated lay-up. The agreement for the matrix dominated lay-up was not as close. Further testing would be required to determine if this discrepancy is caused by test scatter, or inaccurate predictive methodology.

The air-to-ground baseline spectrum was found to increase life, and the increase was accurately predicted for the matrix dominated lay-up. Life was over-predicted for the fiber dominated lay-up. Further testing would be required to determine if this discrepancy is caused by test scatter, or predictive methodology.

The predictive methodology that was used is capable of estimating the effects of large changes in spectra, but improvement is desirable in areas as previously noted. Some of the discrepancies in test and prediction, e.g. truncation to 70% test limit stress, are probably a result of test scatter. Further testing, and perhaps improvements in test techniques, is required to better understand the effects of spectra variations on life. Improvements in test techniques could include monitoring of bending strains caused by eccentric loading of the specimens, and development of techniques to minimize these bending strains.

2. TEST RESULTS AND PREDICTION CORRELATIONS - Detail test results are tabulated in Appendix B, including specimen identification and geometry, load level, and life.

a. Constant Amplitude and Static Strength - Figure 32 summarizes constant amplitude test results. These correlations of test lives obtained in the methodology development program (Section V) and test lives obtained in the spectra variation test program substantiate the validity of the constant amplitude data used in making spectrum predictions.

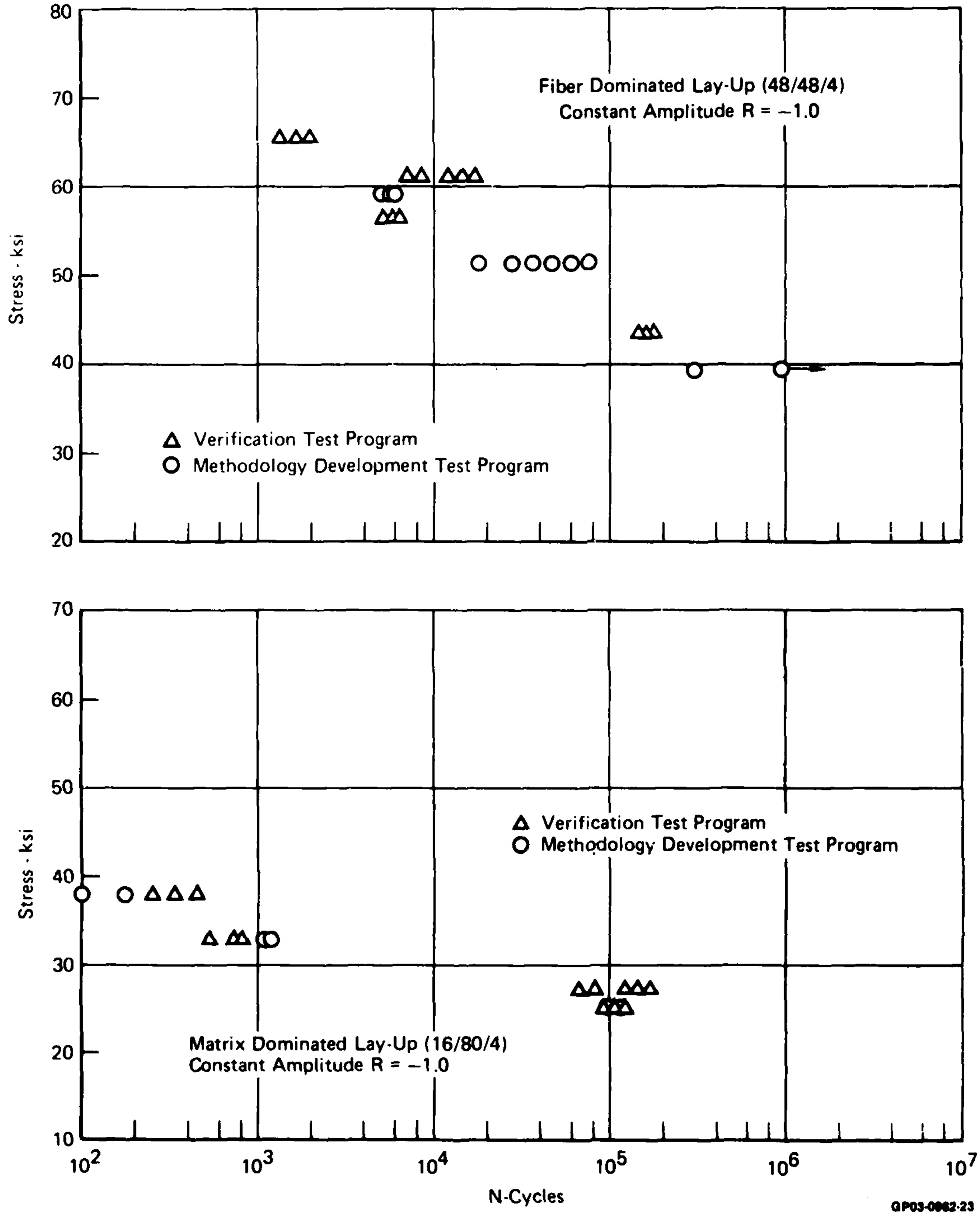

Figure 32. Constant Amplitude Fatigue Lives

Table 7 summarizes the measured static strength of the test specimens. These strengths were greater than those measured in the methodology development program (Table 4), especially for the fiber dominated lay-up. The cause for these differences was not determined.

TABLE 7. STATIC TEST RESULTS

Test Condition	Layup			
	48/48/4		16/80/4	
	Load (lb)	Stress (KSI)	Load (lb)	Stress (KSI)
Compression	37,900	△1 97.18	21,700	55.64
	37,600	△1 96.41	21,300	54.62
	34,960	89.62	21,100	54.10
	33,750	86.54	20,850	53.46
	32,850	84.23	20,300	52.05
Average	35,410	△1 86.80	21,050	53.97
Tension	30,400	77.95	16,100	41.28
	29,400	75.38	16,000	41.03
	29,300	75.13	15,950	40.90
	29,150	74.74	14,700	37.69
Average	29,560	75.80	15,688	40.23

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Note: △1 The higher values of 97.18 and 96.41 ksi were obtained in tests in different equipment, and were excluded from the test average.

b. Baseline Spectrum and Test Limit Stress Variations - Predicted and test lives for the baseline spectrum are presented in Figure 33. The baseline spectrum is the F-15 measured mix, truncated to 55% test limit stress. The value of test limit stress used in the subsequent variation tests was 72.5 ksi for the fiber dominated lay-up, and 37.5 ksi for the matrix dominated lay-up.

Also shown in Figure 33 are the results for the variations in test limit stress. The baseline test limit stress for the fiber dominated lay-up was 85% of F_{cu}, and variations of 90% and 80% of F_{cu} were evaluated. The baseline test limit stress for the matrix dominated lay-up was 66.7% of F_{cu}, and variations of 85% and 55% of F_{cu} were evaluated. In this figure, and in the following figures, the test mean is indicated by an arrow symbol. The value of test mean was estimated by Weibull statistical analyses, described in Section VI.3. The value of test mean and predicted life curve for the baseline spectrum is repeated on each subsequent figure summarizing test and prediction for spectra variations.

The mean lives shown in Figure 31 for the $\pm$ 5% test limit stress variations were estimated by interpolation of the data presented in Figure 33. The results summarized in Figure 31 indicate the effect of a 5% increase in test limit stress is to reduce life by a factor of three, and the effect of a 5% decrease in test limit stress is to increase life by a factor of two. These effects are generally the same for both the fiber and matrix dominated lay-ups, and were predicted with reasonable accuracy.

c. Clipping to 90% Test Limit Stress - In this variation, all loads are retained, but those exceeding 90% TLS are reduced to that value. Results are summarized in Figure 34. The predicted effect is to increase life by approximately 60% (87% for the fiber, and 32% for the matrix dominated lay-up). The test indicated a smaller effect, approximately 12% (15% for the fiber and 11% for the matrix dominated lay-up).

d. Addition of Overloads - In this variation, an additional load was added, once every 1000 spectrum hours. The value of the load was 115% TLS for the fiber dominated lay-up, making the maximum load 97.75% of F_{cu}. For the matrix dominated lay-up, the value of the overload was 125% TLS, making the maximum load 83.33% of F_{cu}. Results are summarized in Figure 35. The predicted effect in the fiber dominated lay-up was to reduce life by a factor of 39, the test value was a factor of 47. The predicted effect in the matrix dominated lay-up was to reduce life by a factor of 2, the test value was a factor of 24. The cause for the discrepancy in test and predicted life for the matrix dominated lay-up has not been determined. Further prediction methodology development is required to accurately predict the effects of overloads.

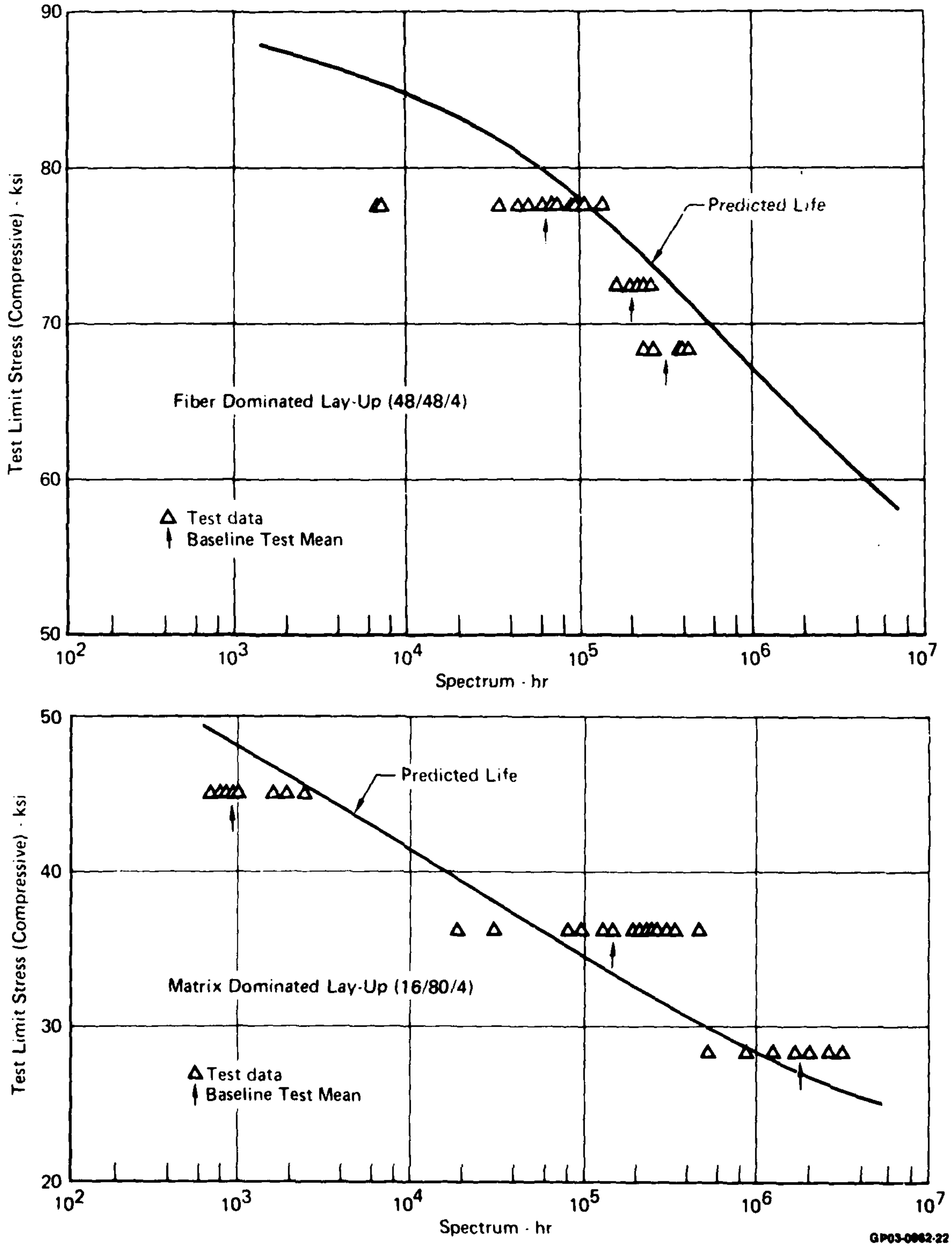

Figure 33. Baseline Spectrum - Measured Mix, Truncated

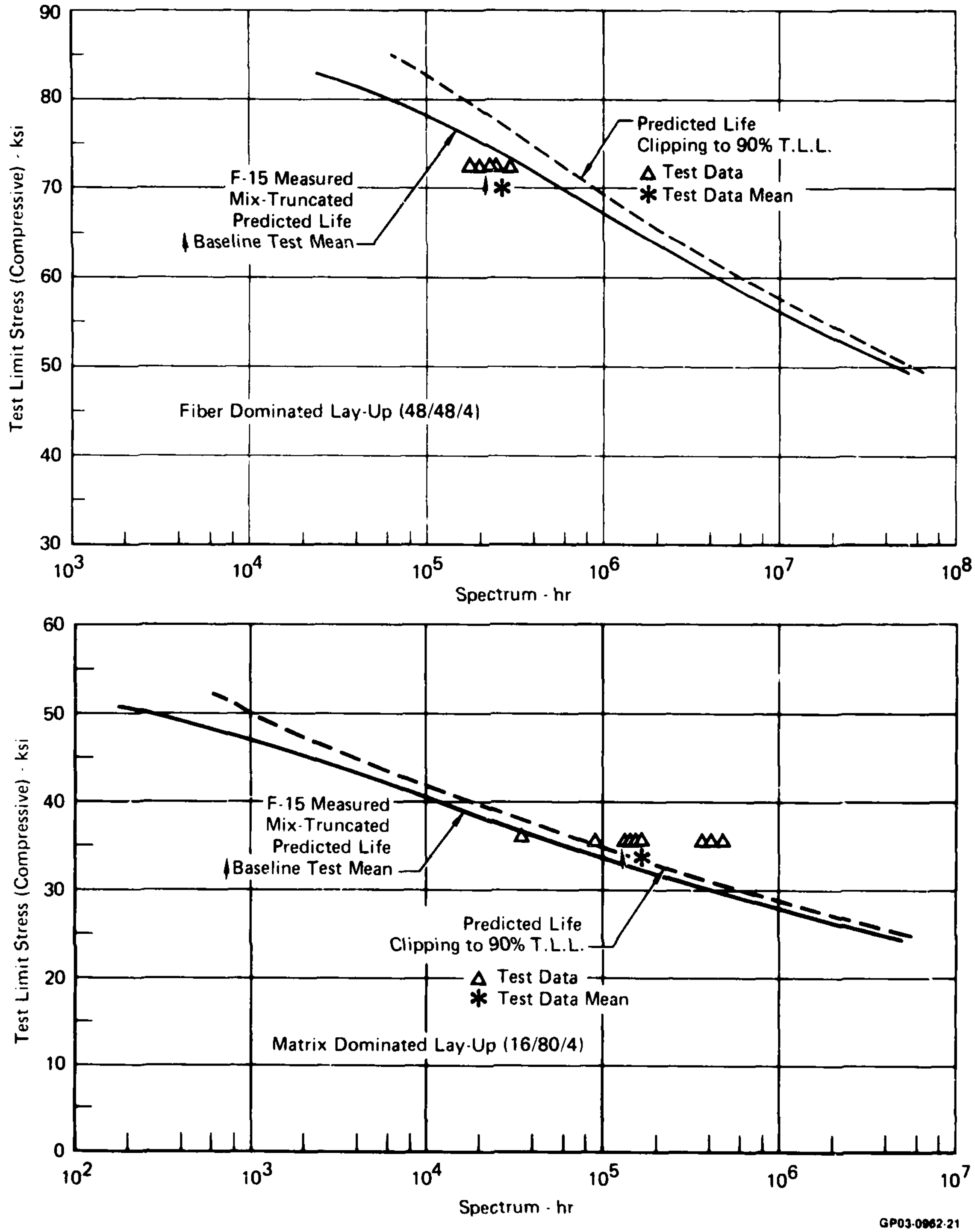

Figure 34. Clipping to 90% Limit Stress

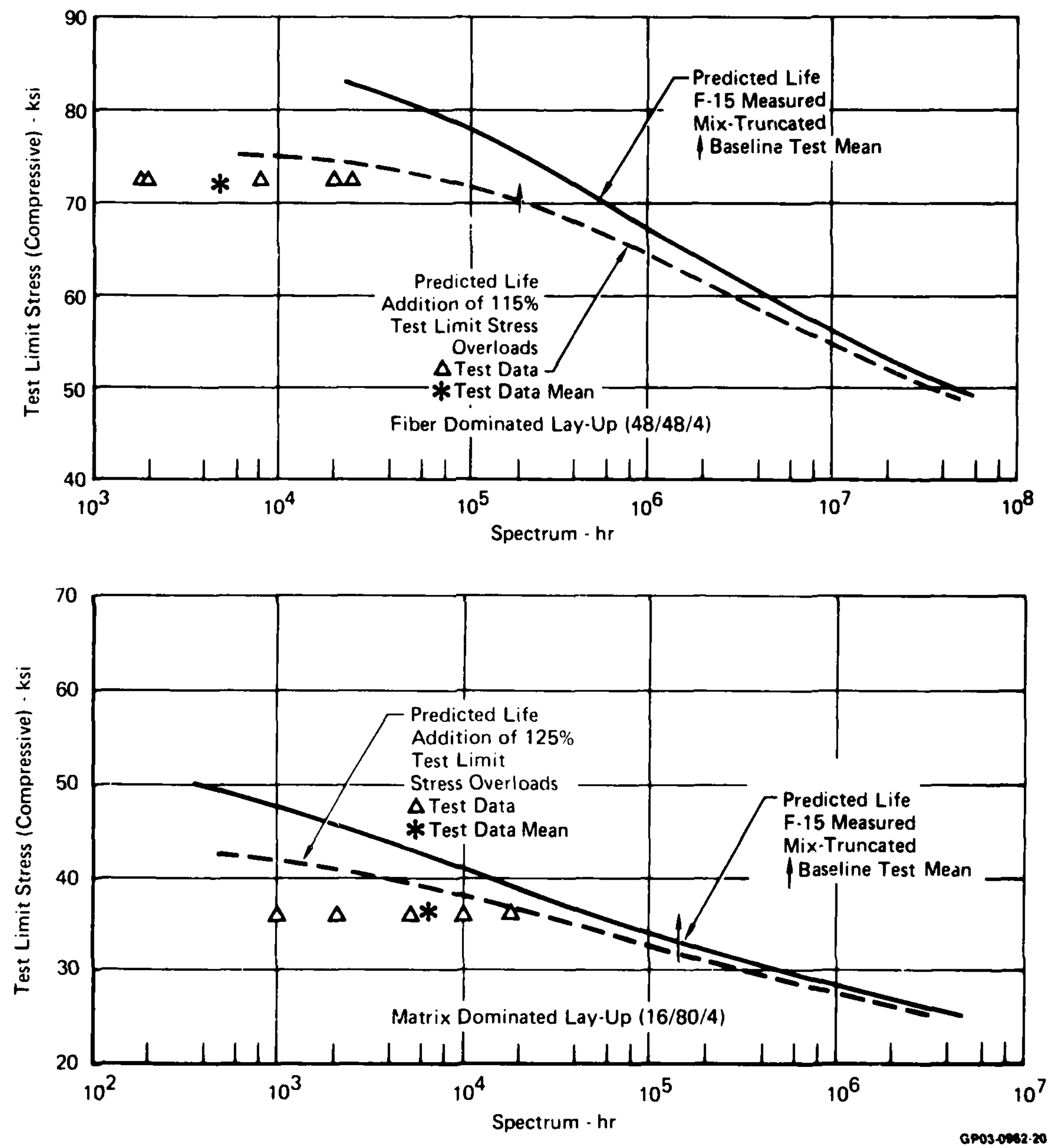

Figure 35. Addition of Overloads.

e. Original Measured Mix - In the baseline spectrum used in the majority of testing, the loads below 55% TLS were removed for test economy. This level of truncation reduces the number of loads from 18,000 to 5,000 per 1000 hours. The effect of this truncation was evaluated by performing tests with the entire spectrum, results are summarized in Figure 36. The predicted effect of adding the low loads was to reduce life by 1%, for both layups. In test, the life of the fiber dominated lay-up was reduced by 48%, and matrix dominated life was reduced by 40%. The consistency of the test data would seem to indicate that the predictions are in error, and that the effects of low loads are to reduce life more than estimated with simple analyses. However, the results of the next test series (Truncation to 70% Test Limit Stress) makes this tentative conclusion subject to question.

f. Truncation to 70% Test Limit Load - To further evaluate the effects of removal of low loads, those with values less than 70% TLS were removed. This reduces the number of loads from 5,000 to 1,000 per 1,000 hours, as compared to the baseline spectrum. The predicted effect was to increase life, 25% for the fiber dominated lay-up, and 19% for the matrix dominated lay-up. However, test results (Figure 37) indicate life was reduced: 30% for the fiber dominated lay-up, and 42% for the matrix dominated lay-up. This discrepancy of prediction and test has not been explained, but these results place doubt on the conclusions tentatively obtained in addition of low loads variation. It cannot be logically concluded that both addition of low loads, and removal of low loads cause reduction in life, as indicated by the test data.

g. Clipping of Tension Loads - The baseline spectrum is primarily compression dominated, with lesser tension loads. The effect of these tension loads was evaluated by removing them, clipping their values to zero. There was expected to be small effect of this clipping, and the predicted effect was to reduce life. This predicted effect is a result of the stress ratio correction used in the analysis (Figure 23) which for small values of tension minimum load is apparently slightly unconservative, i.e., life is overestimated. When these small values of tension are reduced to zero, life is predicted to be decreased. With tests using the fiber dominated lay-up, there was little effect of removing the tension loads Figure 38. Life was slightly reduced, where logically life should be slightly increased. With the matrix dominated lay-up, there was a dramatic decrease in life, by a factor of six. This result is in disagreement with expected behavior, and test results for the fiber dominated lay-up. Further prediction methodology development is required to accurately predict the effects of tension loads.

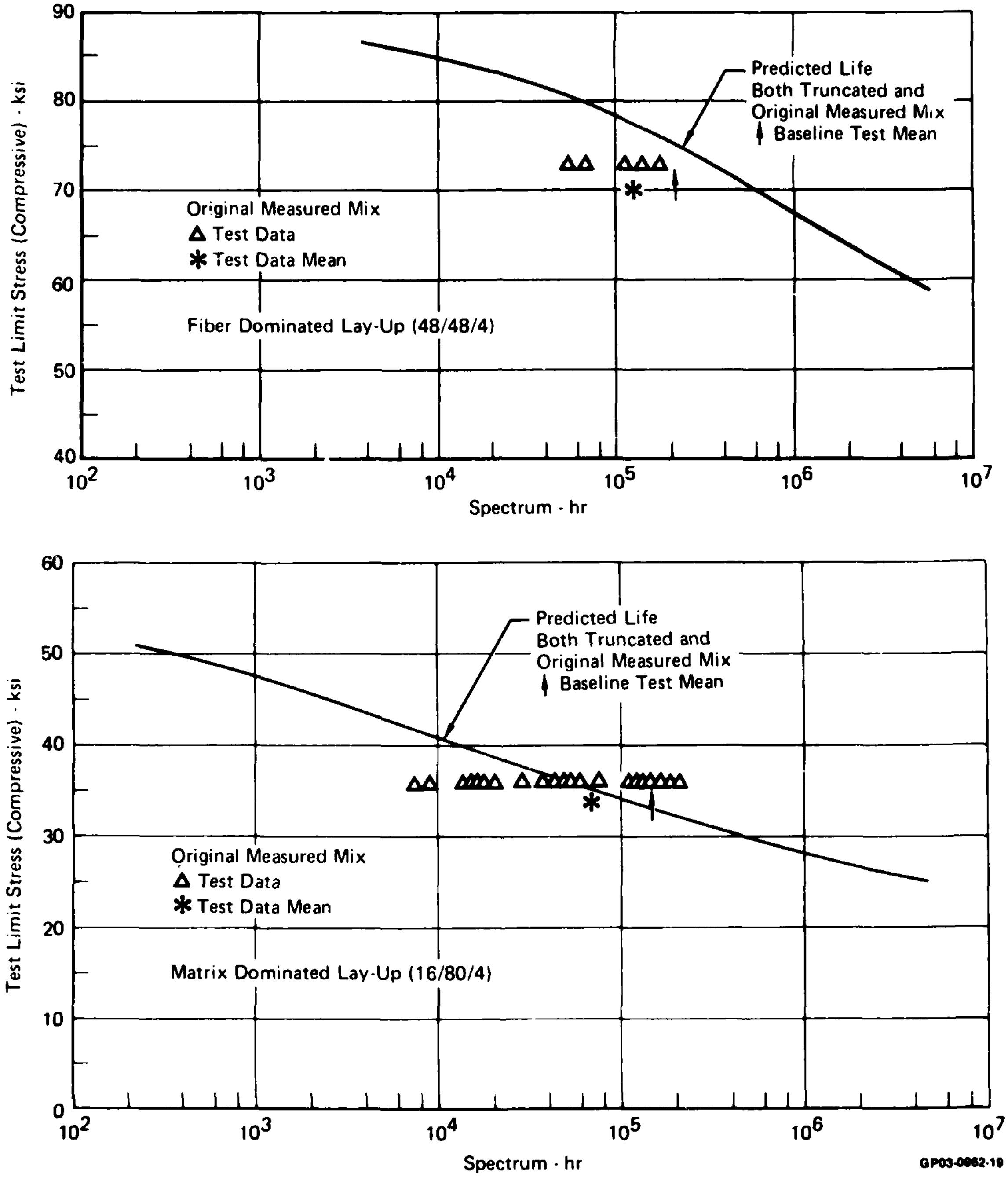

Figure 36. Original Measured Mix

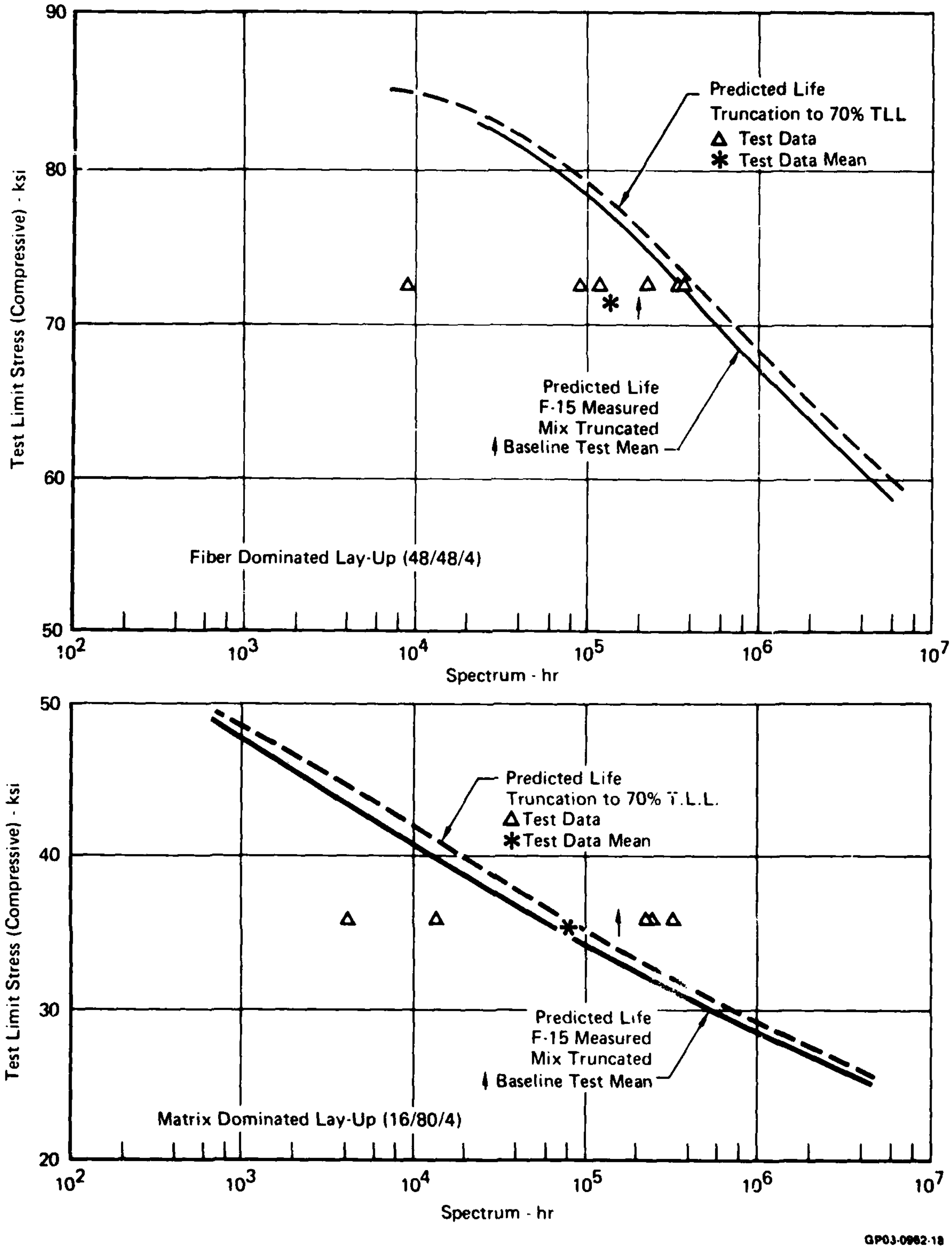

Figure 37. Truncation to 70% Test Limit Stress

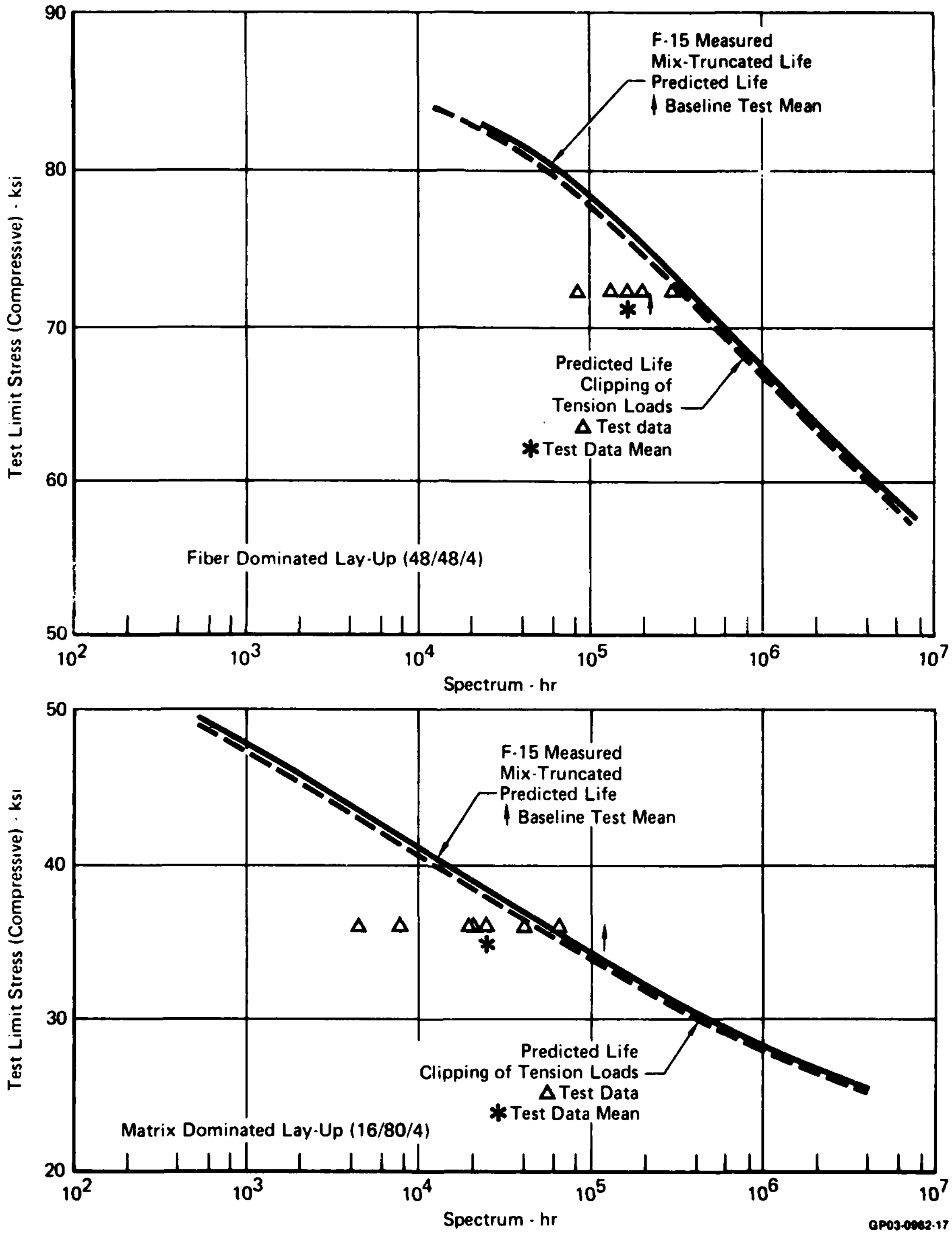

Figure 38. Clipping of Tension Loads

h. Increased Severity and Number of Air-to-Air Loads - The effects of usage changes on life were evaluated by testing with a more severe spectrum developed by increasing the magnitude and value of the air-to-air loads. As expected, life was significantly decreased (Figure 39). With the fiber dominated lay-up, test life was reduced by a factor of 100. The predicted decrease was a factor of 155. With the matrix dominated lay-up, test life was reduced by a factor of 27. The predicted decrease was a factor of 6.

i. Air-to-Ground Baseline - Usage changes were also evaluated by testing with a benign spectrum, the Air-to-Ground baseline. This spectrum contains fewer loads, of lesser magnitude, than the baseline spectrum. The test life increase for the fiber dominated lay-up was a factor of 3.4 greater than the baseline life, while the predicted factor was 7.4. The corresponding values for the matrix dominated lay-up were a test life increase of a factor of 6.0, and predicted increase of a factor of 5.0. The results are shown in Figure 40.

3. STATISTICAL ANALYSIS - Analyses of each data set were made using a procedure based on the Weibull distribution:

$$F(t) = 1 - e^{-(x/\theta)^{\beta}}$$

where: $F(t)$ = cumulative probability of failure
β = Weibull slope
θ = characteristic life
x = life, spectrum hours

The values of β and θ were computed using a least squared regression analysis of the data. The techniques used to estimate these values are outlined in Reference 7. Comparisons of cumulative probability and life for each of the spectra variations are shown in Appendix C. The 90% confidence limits shown in Figures 29, 30 and 31 were derived from the overall average standard deviation of log life (0.477), and characteristic lives summarized in Tables 5 and 6.

The range of Weibull slopes in Tables 5 and 6 is from 0.56 to 6.91. It is hypothesized that all of these data should be from the same statistical population in terms of scatter (Weibull slope). Weibull slopes computed from the data are sample values and as such are random variables. These random variables have their own theoretical probability distributions and should exhibit scatter according to those distributions. One of these probability distributions is the χ^2 distribution of standard deviation, σ. The standard deviation of $\log_{10}$ (life) is related to Weibull slope through the equation

$$\sigma = \frac{0.5570}{\beta}$$

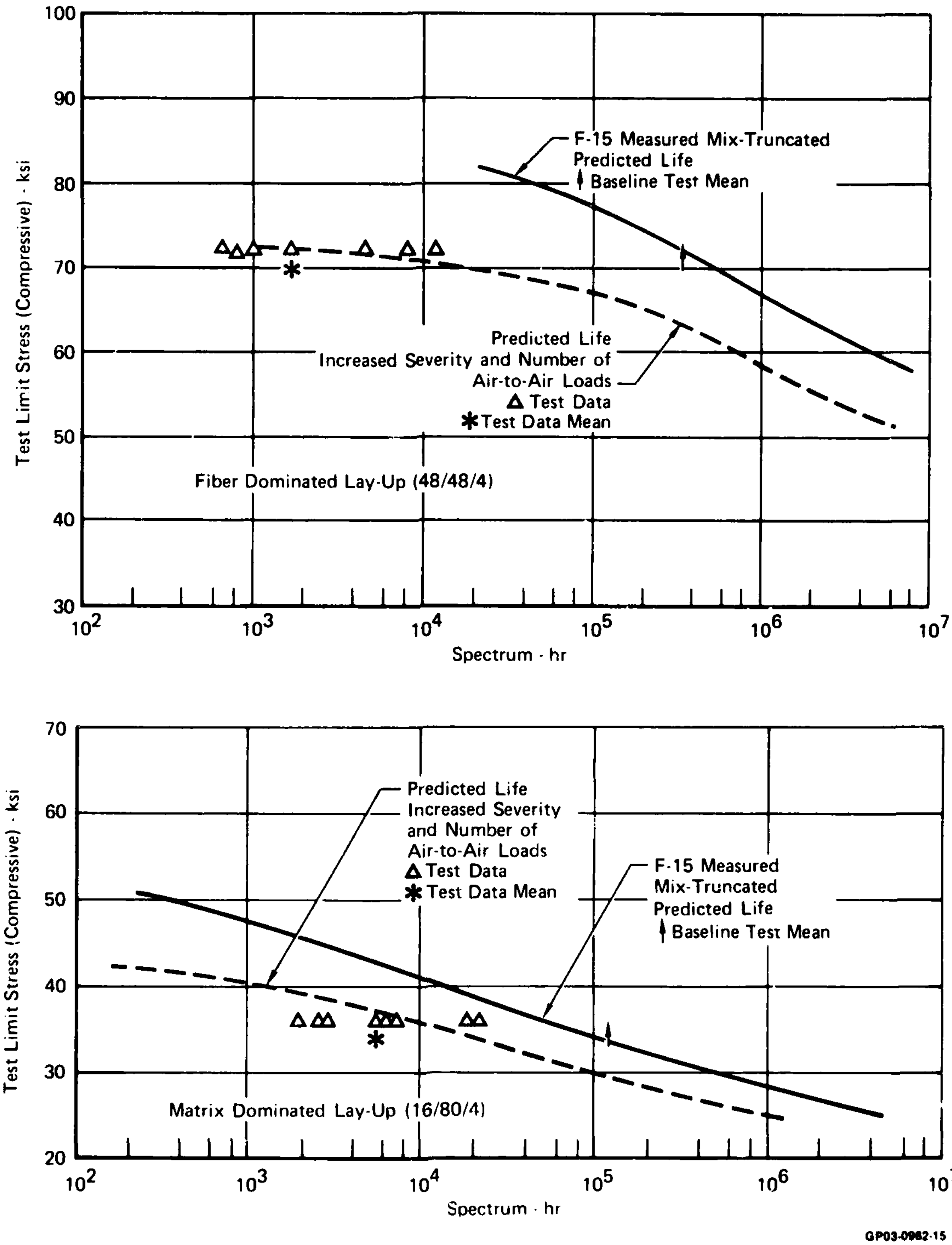

Figure 39. Increased Severity and Number of Air-to-Air Loads

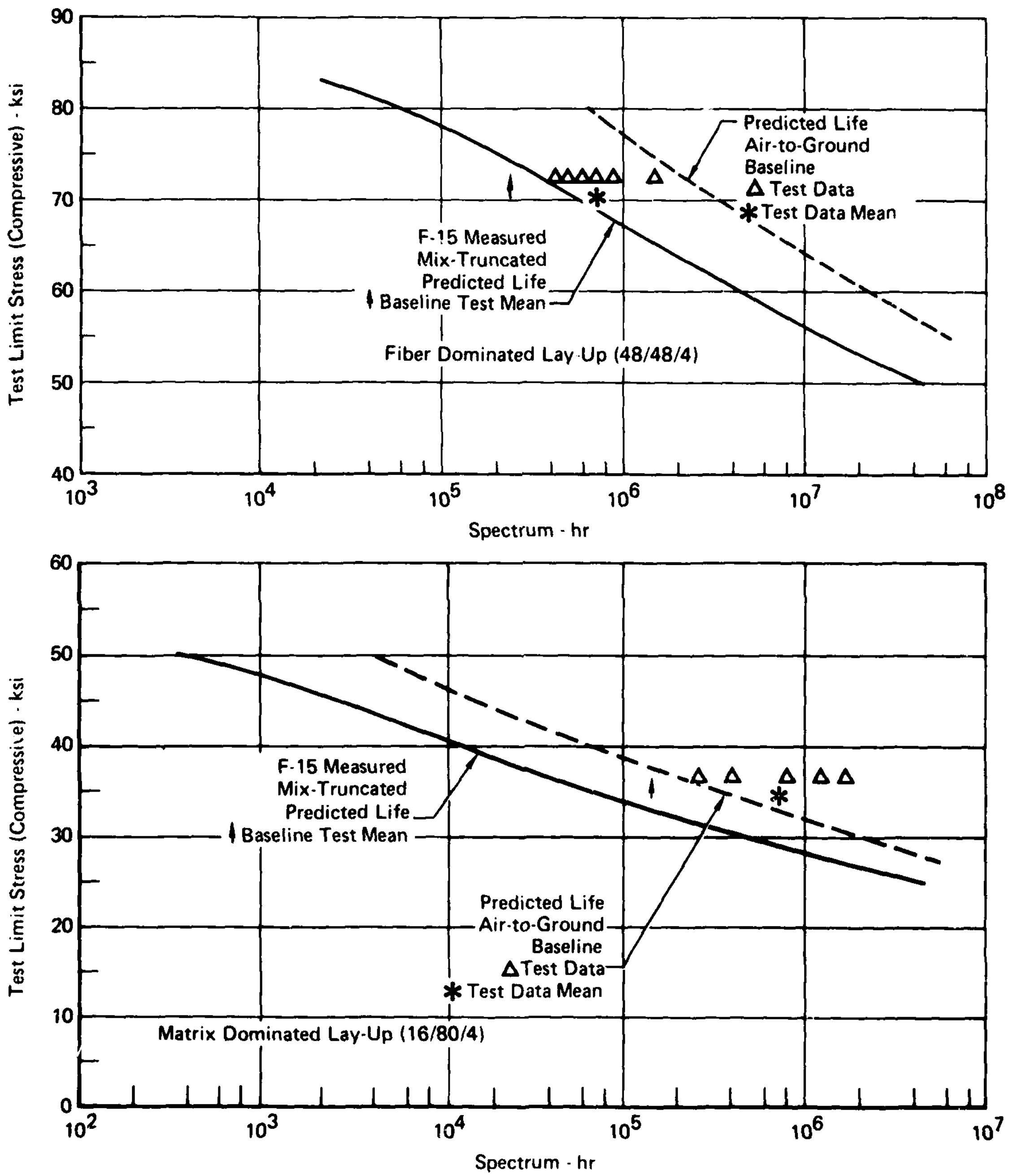

Figure 40. Air-to-Ground Baseline

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Figure 41 presents a comparison of the theoretical distribution of sample Weibull slope with the measured distribution. Figure 42 presents a similar comparison of sample and measured distributions of standard deviation of log life. These two comparisons indicate the ranges of Weibull slopes and standard deviations is greater than would be theoretically expected. Comparisons of theoretical and measured scatter for aluminum data are presented in Figures 43 and 44, demonstrating that scatter can occur in the manner theoretically expected.

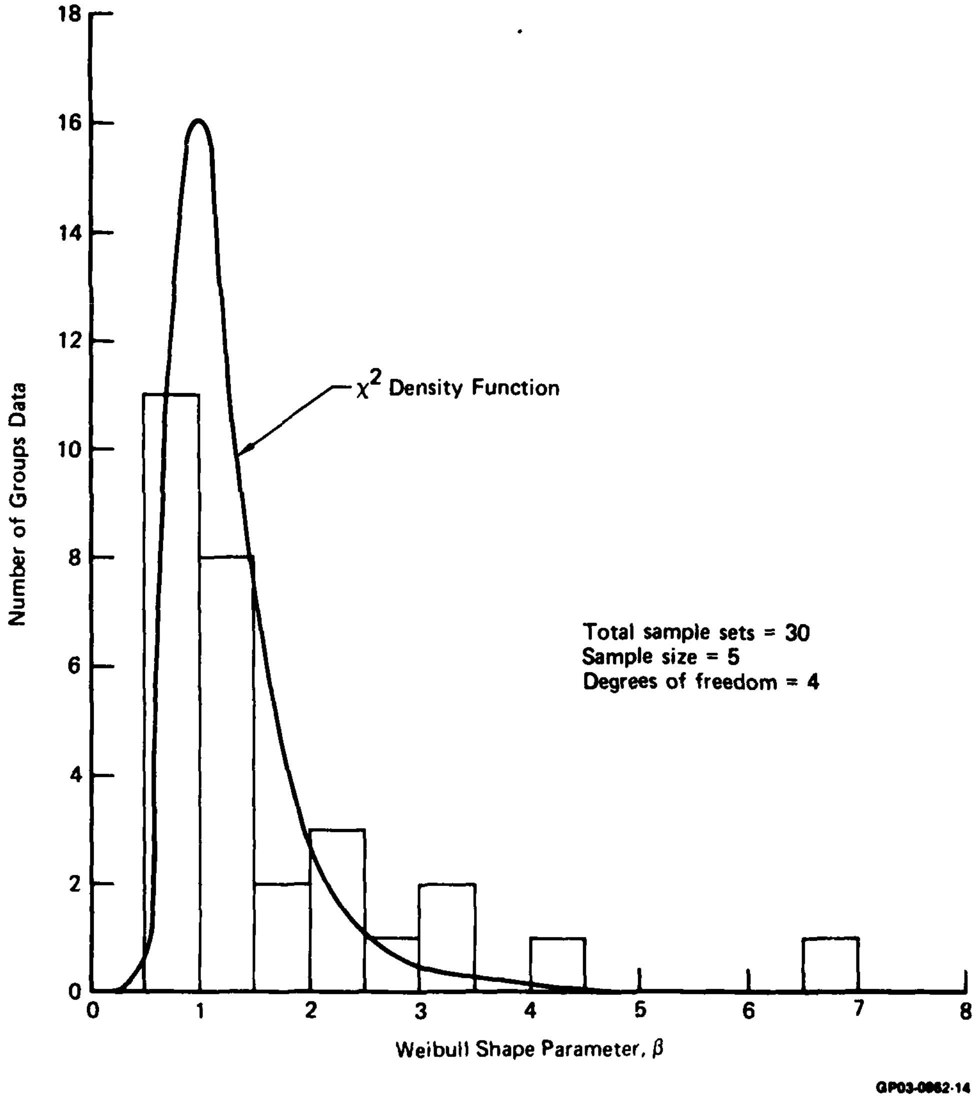

Figure 41. Theoretical and Observed Distribution of Weibull Shape Parameter

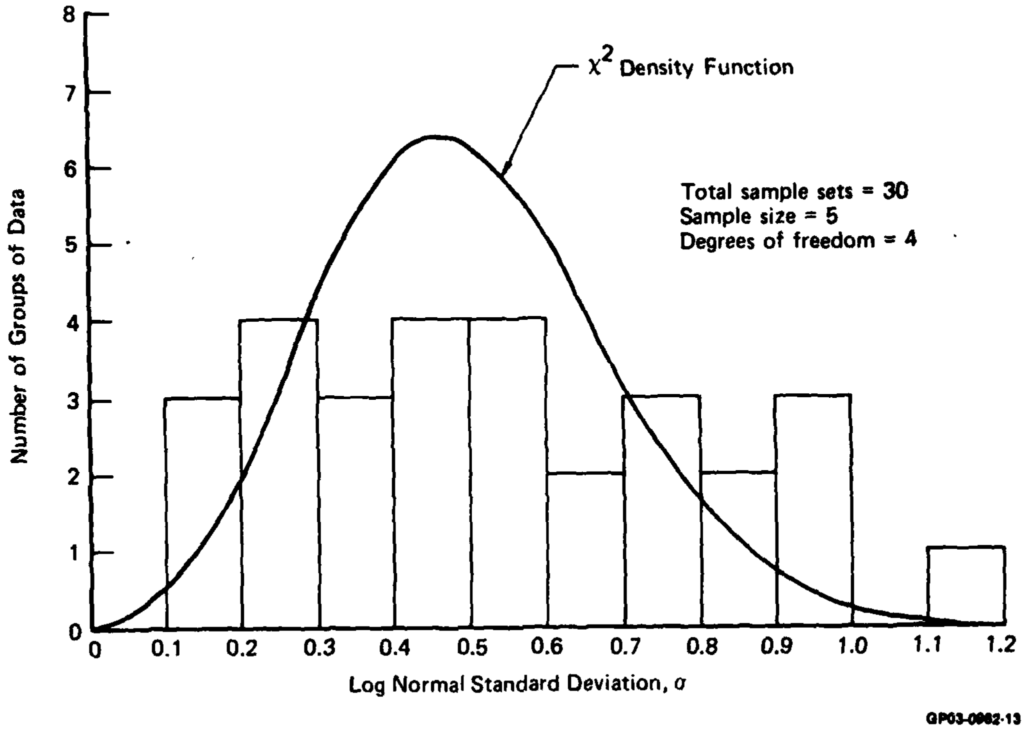

Figure 42. Theoretical and Observed Distribution of Log-Normal Standard Deviation

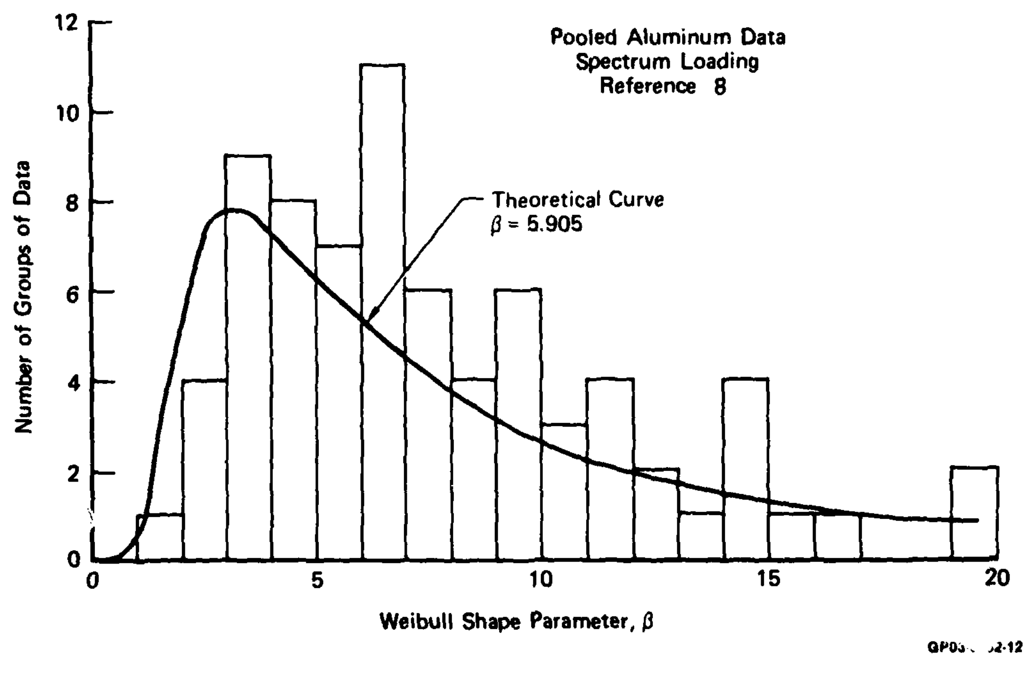

Figure 43. Distributions of Weibull Shape Parameter for Aluminum Data

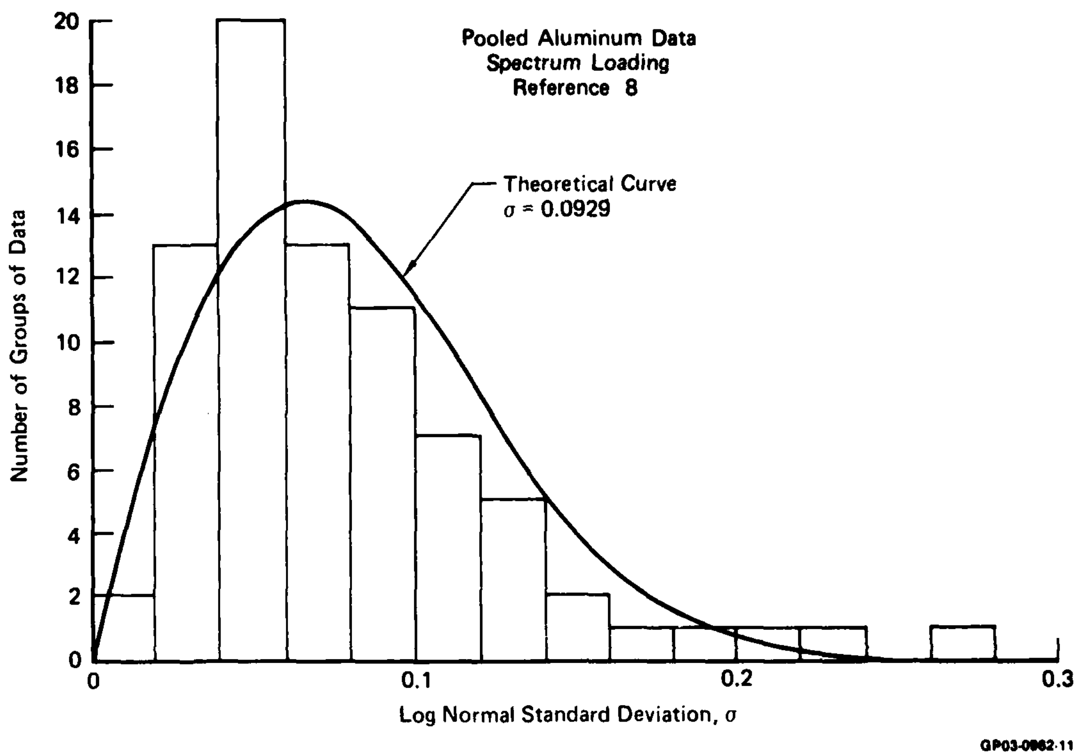

Figure 44. Distributions of Log-Normal Standard Deviation for Aluminum Data

The cause of the unexpected range of scatter exhibited by the composite specimens has not been determined. It is conjectured that the cause of the scatter contributed to the sometimes poor correlation of predicted life and test life, and the sometimes unexpected effect of spectrum variations on test life. Early testing in this program led to the expectation that the scatter would be similar to that observed in metallic testing. In the work reported in Reference 8, the average log-standard deviation of log-life was found to be 0.100, and the Weibull slope was found to be 5.27. The comparable values for tests of this program were a log standard deviation of 0.477 which corresponds to a Weibull slope of 1.209. Thus, the average scatter was approximately five times as large as for the metallic specimens. As shown in Tables 5 and 6, the ratio of predicted to test life ranged from 0.33 to 4.66, which is a large range in prediction accuracy.

An example of an unexpected effect of spectrum variation upon test life is the truncation to 70% Test Limit Stress variation using the matrix dominated lay-up (Table 6). This variation was expected to increase life from the baseline (predicted life ratio = 1.19). The test results indicated a reduction in life (test life ratio = 0.58).

Further testing would be required to determine the causes of the scatter demonstrated in test, and to investigate the unexpected test results that sometimes occurred.

SECTION VII

SUMMARY AND RECOMMENDATIONS

1. SUMMARY - The impact of spectrum variations upon fatigue life is summarized in Figure 31. The measured lives are normalized with respect to the baseline life, the life developed with the F-15 Measured Mix-Truncated. The variations found to have the greatest effect were those that increased the magnitude or frequency of high loads in the spectrum. Two variations, Addition of Overloads, and Increased Severity and Number of Air-to-Air Loads, caused more than an order of magnitude reduction in life. A 5% increase in test limit stress caused a 60% decrease in life. The last variation that modified the high loads was clipping to 90% Test Limit Stress. This variation caused small change in test life - an average of less than 15% increase in life.

Variations predicted to have small impact on life were those that change the lesser loads in the spectrum. These variations are the Addition of Low-Loads to Match Original Measured Mix, Truncation to 70% Test Limit Stress, and Clipping of Tension Loads. Test lives for these variations were somewhat less than predicted. However, the test lives were more than 75% of baseline life, with the exception of life with the matrix dominated lay-up subjected to the Clipping of Tension Loads spectrum. This test result is in disagreement with prediction, and the test result for the fiber dominated lay-up. The apparently low value of test mean may be a result of scatter, demonstrated in this test series and the others in the program. As expected, the Air-to-Ground spectrum, and a 5% decrease in test limit stress both increased life.

Composite laminates have excellent fatigue properties. As a result, the stress levels used in the test program were higher than would typically be used in aircraft. Use of typical stress levels would have resulted in extremely long test times, and lives much greater than of interest in fighter aircraft. It is believed the conclusions developed from the tests of this program are valid, even though the stress level increase was required.

2. RECOMMENDATIONS FOR FORMULATING SPECTRA - Spectra variations generated in this program represent the exceedance content of loads applied to the upper wing skin of a fighter aircraft, therefore the recommendations for spectra formulation is restricted to such applications. Figure 31 summarizes the effect of the studied variations upon fatigue life. As noted previously, the variations which increased the frequency or magnitude of the high loads in the spectrum had the greatest impact on life, and the effect was to reduce life. This is in contrast to the effects of overloads in metal structure. In metals, discrete overloads can significantly increase life. In Reference 1, adding single high loads was found to increase metallic crack growth life to nearly 650% of the baseline for the most extreme case.

Thus, high loads in the spectrum exceedance are very important in determining composite fatigue life, and care should be exercized in their selection to insure that they represent expected service usage. High loads are extreme values of the probability distribution of peaks. Accurate prediction of extreme values is more difficult than prediction of mean values or moderate probability values and greater reliance on judgement will be required.

Figure 31 shows that test limit stress can significantly affect life. The 10% stress level variation (± 5%) summarized in that figure causes about a 500% change in life. In Reference 1, a 10% stress level variation caused only about a 50% change in crack growth life of metal structure. Thus, it appears that composite fatigue life is much more sensitive to stress level than metal fatigue life. Stress levels in full scale structure are dependent upon the structural arrangement and external loads, and limit stress is not an independent variable. In the design phase of an aircraft system, element tests are used to aid in the selection of design limit stress and structural sizing, and the stress level is an independent variable. In the selection of stress levels, and the selection of number of replications for each test condition, the sensitivity of life to stress level, and the large scatter evidenced by composite structure, needs to be considered.

Development of fatigue load spectra for a fighter aircraft consists of the following steps:

(a) Missions are defined. In the design stage, these definitions are based on expected utilization. For aircraft fleets in service, these descriptions are based on fleet usage tracking, developed from flight logs, counting accelerometers, or multi-channel recorders. Mission types are air-to-air, air-to-ground, and instrumentation and navigation. The number of each of these missions is estimated by operational analysis for aircraft in the design stages, and by fleet usage tracking for service aircraft.

(b) Each of the missions is subdivided into mission segments: ascent, cruise, loiter, combat, and descent. For each of these mission segments, the time expended, altitude, aircraft weight, Mach No., and load factor exceedance data are estimated. These estimates are based on operational analyses of the aircraft, guided by fleet usage tracking of that aircraft fleet or similar older aircraft. Exceedance data are shown schematically in Figure 1. Air-to-air and air-to-ground combat segments are the most severe, and greatest attention is directed toward estimating weight, Mach No., and altitude during these segments.

(c) In order to generate a spectrum for analysis and test of structural elements, it is necessary to convert load factor to stress for selected locations on the aircraft.

External loading distributions are computed for each of the mission segments. These external loads are determined as a function of aircraft weight, Mach No., altitude, and load factor. The values of these parameters are those estimated in step (b). These external loads computations should consider the dynamic response behavior of the elastic aircraft. Internal loads for each of the external loads distributions are then computed. The resulting relationships of load factor to stress are summarized, e.g., Figure 6.

(d) The summaries of load factor to stress relationships (step (c)) are used in combination with the mission segment load factor exceedances estimated in step (b) to generate internal load and stress exceedance data sets. These exceedances for each mission segment are then summed to create internal load and stress exceedances for each mission type. The resulting summaries of exceedances are depicted in Figure 7.

(e) The next step is generation of cycle-by-cycle stress sequences for each mission type. The stress exceedance curves depicted in Figure 7 represent an important part of the fatigue spectrum generation process. These define the number of peaks and valleys in the fatigue spectrum. The method of combining peaks and valleys to form the stress time history needed for analysis and test is important for metals. The method is less important for composites. Composite life is more controlled by the high loads in the spectrum than the lesser loads; hence, the sequence of lesser and high loads is less important. However, simplification of random cycle-by-cycle, flight-by-flight realistic sequence spectra should be minimized. Such simplifications can cause uncertainties as to the applicability of the test results to service aircraft. Therefore, the techniques developed for metal structure should be followed, as outlined below.

(1) Make the length of the spectrum repeatable sequence approximately one-tenth of the test life requirement. Load interaction effects on composite life have not been fully investigated. The results of this program indicate that high loads decrease life, and in the absence of complete understanding of their effects they should be introduced in a uniform manner throughout life. Multiple repeats of the spectrum sequence assures this uniform distribution is attained.

(2) Use random valley and peak coupling, using procedures similar to those outlined in Reference 2, to combine the loads for each mission type into a sequence of random loads.

(3) Use of the procedures outlined above will result in sequences of loads for each mission type. It is not necessary to sequence the loads in each mission type by mission segment, i.e. by separately sequencing loads for ascent, cruise, loiter, combat, cruise, and descent. The work described in Reference 1 indicated that the sequence of loads within a flight had small impact on metallic crack growth life. With the lesser effect of low loads upon life of composites, as compared to metals, the sequence of low loads is relatively unimportant. The sequences of loads in each mission type are next divided into individual flights, by the introduction of loads that simulate ground (landing) loads.

(4) In order to eliminate cycles which do not contribute to degradation, the cycles which have peaks less than 50% of limit stress can be eliminated.

(5) Retain the tension loadings. The predicted and test effect of tension upon life, as determined in this program, leaves uncertainties as to the effects of tension. Because of these uncertainties, the tension loads should be retained in the test spectrum.

(6) Combine individual flights from the different mission types to match the mix of usage estimated in step (a).

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APPENDIX A

INPUT DATA FOR STRESS HISTORY GENERATION AND MODIFICATION

This appendix summarized input data sets used to generate stress histories for the study. The first of these two sets of data are used in a computer program (Reference 2) to generate stress histories using random noise theory. A Gaussian random time history is generated and searched to detect peaks and valleys. The Gaussian peak and valley history is transformed to simulate real time history. The resulting list of peaks and valleys is filtered, using input rise and fall criteria, to eliminate small stress excursions.

The second set of input data is used in another computer program (Reference 3) that accepts lists of filtered peaks and valleys, generated by the first program, along with control information. This second computer program combines and creates variations of the input lists.

Both of these computer programs were used in a study of load sequence effects on crack growth, summarized in Reference 1.

1. INPUT DATA - STRESS HISTORY SIMULATION

a. Gaussian Time History Generation - The program used to generate the time histories is described in Reference 2. Input consists of the number of time points to be computed, power spectral density (PSD) as a function of frequency, rise and fall criteria to be used with cycle counting, and an initial number to be used in a random number generator. The input data format is described in the computer program listing, via comment cards, Appendix A of Reference 2. Example input is presented in Appendix B.

The power spectral density is usually input as (% Design limit stress)^2 vs. frequency, cps. Program input includes the number of points to be input, and the frequency and PSD value for each of the input points. Experience has shown that many peaks and valleys will be missed if the upper limit of the PSD plot is not artificially extended with zero values for PSD, as depicted in Figure 1. The time represented by the input values of PSD upper frequency and the number of time points to be computed is,

$$t = \frac{N}{\omega_u}$$

By increasing ω_u artificially, the time t is reduced for a given number of points, N, effectively increasing the number of points per unit time and hence per peak. Trial runs have shown that the

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use of NR = 2^{NR} = 4, as defined in Figure A-1, will usually result in adequate sensitivity and peaks and valleys will not be missed.

The subroutine within the program that computes the Gaussian time history requires as input a value of PSD for each time point computed. The number of time points to be computed, typically 32,768, is usually much larger than required to adequately define the shape of the PSD plot; hence, constant values of PSD are used in ranges as shown in the example of Figure A-1. In that example, the number of time points is 128 and the frequency scale has been multiplied by four to increase the sensitivity of the output. Therefore, 128/4 = 32 values of PSD are required in the computation of the A(K) terms. In this example, 8 constant values of PSD are used to represent the 32 required values.

b. <u>Mapping of Gaussian Process to Real</u> - The exceedance curve for a Gaussian random time history is symmetrical with zero mean, as depicted in Figure A-2. Actual exceedance curves for fighter aircraft are asymmetrical with a non-zero mean, also depicted in Figure A-2. Gaussian random histories are adjusted so that exceedance curves for the Gaussian process and the actual exceedance curves are matched. This is accomplished by using the transformations:

$y = (x + M)^R$ for positive $x + M$

$y = -(-x - M)^R$ for negative $x + M$

and conversely,

$x = y^{1/R} - M$ for positive y

$x = -(-y)^{1/R} - M$ for negative y

Transformation of the Gaussian to simulated real time history is controlled thru input of the values, M and R.

Input rise and fall criteria are applied to the transformed peak and valley list. Peaks and valleys that do not meet the required rise and fall criteria are removed from the time history.

Other input to the program controls the size and sensitivity of output occurrence and exceedance tables.

c. <u>Input Data Listings for Study Spectra Load Factor Time Histories</u> - Input data used in generating the load factor time histories analyzed in the study are listed below. Three input data sets were used with this program. The titles associated with the data sets are not part of the input data.

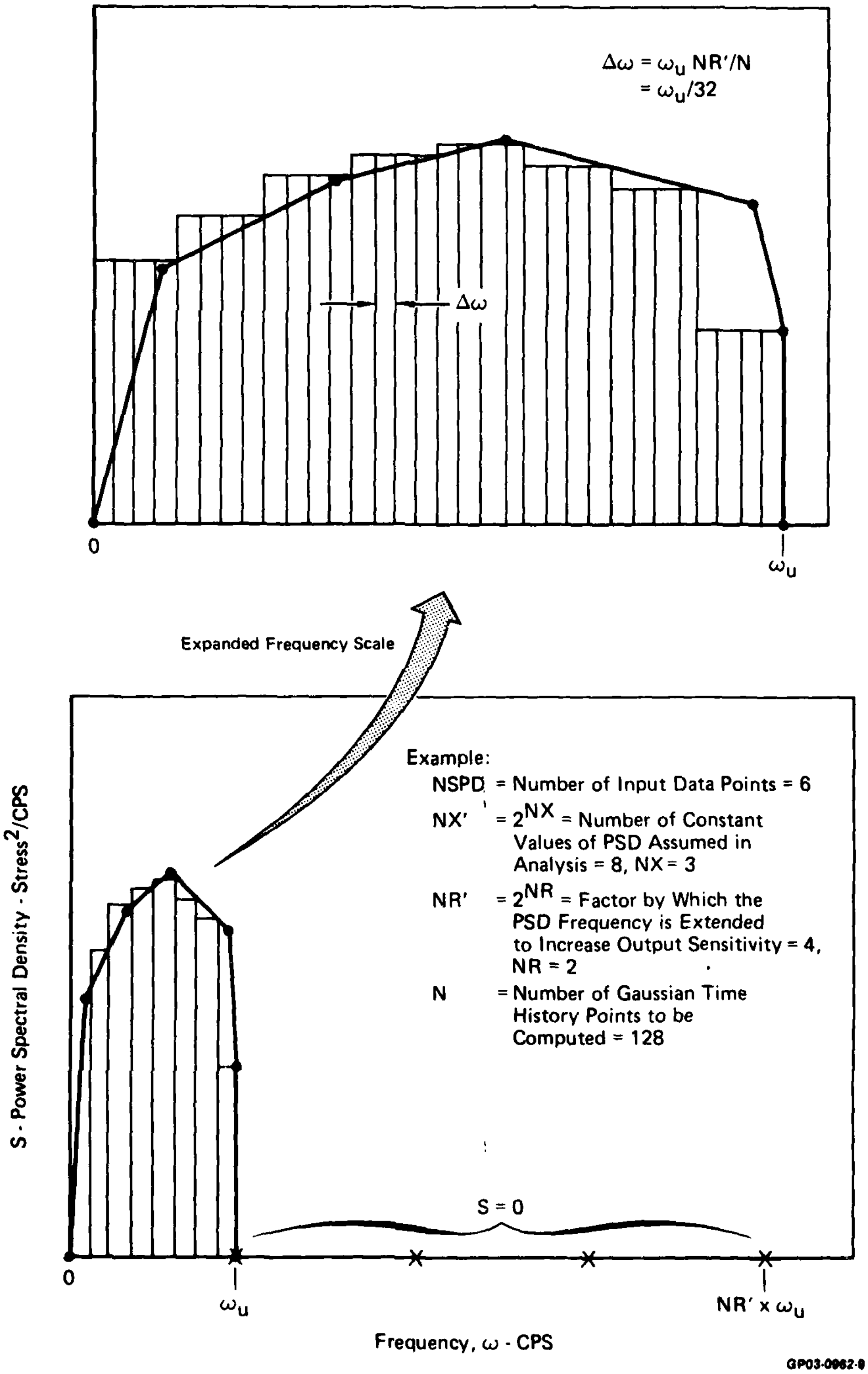

Figure A-1. Example of PSD Input for Random Load History

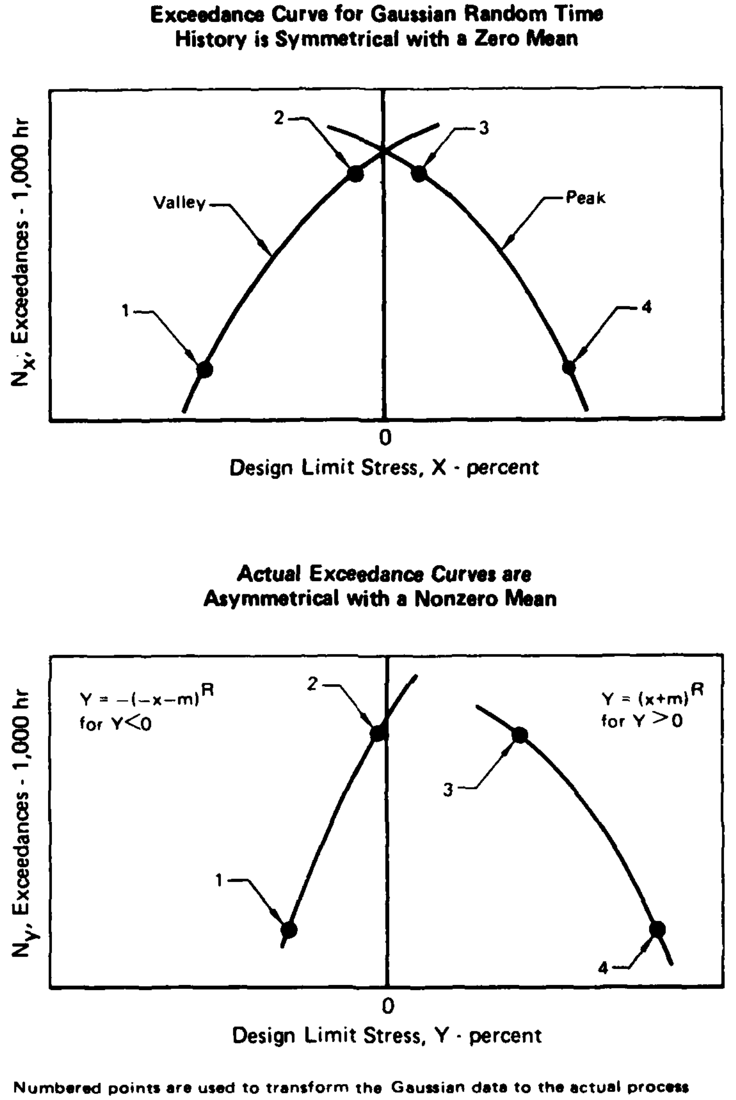

Figure A-2. Comparison of Exceedance Curves

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2. <u>INPUT DATA - MODIFICATION OF STRESS HISTORY SIMULATIONS</u> - A second computer program was used to combine and modify stress history simulations, generated by the first program, and create two baseline spectrum and seven spectra variations. The baseline spectra are Air-to-Ground (A-G) and F-15 Measured Mix-Truncated. The spectra variation types are:

Clipping to 90% Test Limit Stress

Addition of 115% Test Limit Stress Overloads

Addition of 125% Test Limit Stress Overloads

Addition of Low Loads to Match Original Measured Mix

Truncation to 70% Test Limit Stress

Clipping of Tension Loads

Increased Severity and Number of Air-to-Air Loads

Input to the program includes control numbers to create the spectra variations, and data sets of peak and valley stress history simulations obtained from the first computer program. Output includes a sequential list of peaks and valleys, and a table summarizing peak and valley coupling. Input and output are described in Sections 3 and 4 of Reference 3. The input data format is described in the computer program listing, via comment cards, Appendix A of Reference 3. Example input is presented in Appendix B of that reference.

The first output control data are LINE1 and GLØAD. These are used to describe the mission mix (combination of Air-to-Air, Air-to-Ground and Instrumentation and Navigation) and, for each mission type, the stress representing the ground load. The completion of the mission mix description is signaled by input of the value END for LINE1. After completion of the mission mix description, a series of forty-two indicators are read. Combinations of these indicators were used to create the two baseline and seven spectra variations studied.

a. <u>Input Data Listing for Study Spectra</u> - Input data used in generating these baseline and spectra variations analyzed in the study are listed below. Nine input data sets were used with this program.

```
AIR TO AIR BASELINE...
  15    2    7   46  759
 0.         4904.    .0022     3903.     .0044     2561.
 .0067      2240.    .0089     2138.     .0111     1817.
 .0133      1977.    .0156     3126.     .0178     3794.
 .02        3088.    .0222     2093.     .0244     1316.
 .0267      995.     .0289     1303.     .0311     1406.
 .0333      1027.    .0356     802.      .0378     757.
 .04        629.     .0422     443.      .0444     334.
 .0467      379.     .0489     488.      .0511     417.
 .0533      315.     .0556     308.      .0578     276.
 .06        244.     .0622     205.      .0644     154.
 .0667      160.     .0689     160.      .0711     160.
 .0733      160.     .0756     141.      .0778     128.
 .08        116.     .0822     109.      .0844     83.
 .0867      49.      .0889     51.       .0911     60.
 .0933      57.      .0956     47.       .0978     40.
 .1         45.
  12   5.      10.0      10.0      22.2      1.128      1
  13  10.
 800
 9873
 7829
 3649
 4556
 6969
49601
```

```
     AIR-TO-GROUND BASELINE
 15      2       7   101   759
  0.0000        241.      .0008      266.      .0016      221.
   .0024        143.      .0032      121.      .0040      117.
   .0048        117.      .0056      113.      .0064      169.
   .0072        213.      .0080      194.      .0088      219.
   .0096        315.      .0104      396.      .0112      365.
   .0120        251.      .0128      204.      .0136      206.
   .0144        181.      .0152      189.      .0160      219.
   .0168        214.      .0176      201.      .0184      201.
   .0192        202.      .0200      222.      .0208      256.
   .0216        241.      .0224      184.      .0232      160.
   .0240        149.      .0248      124.      .0256      120.
   .0264        131.      .0272      133.      .0280      132.
   .0288        117.      .0296       89.      .0304       81.
   .0312         85.      .0320       98.      .0328      119.
   .0336        106.      .0344       83.      .0352       80.
   .0360         87.      .0368      102.      .0376      102.
   .0384         75.      .0392       55.      .0400       56.
   .0408         65.      .0416       73.      .0424       84.
   .0432         87.      .0440       68.      .0448       61.
   .0456         70.      .0464       66.      .0472       52.
   .0480         48.      .0488       49.      .0496       51.
   .0504         47.      .0512       43.      .0520       48.
   .0528         52.      .0536       47.      .0544       36.
   .0552         28.      .0560       33.      .0568       34.
   .0576         25.      .0584       26.      .0592       34.
   .0600         33.      .0608       24.      .0616       21.
   .0624         21.      .0632       18.      .0640       19.
   .0648         25.      .0656       29.      .0664       25.
   .0672         23.      .0680       27.      .0688       30.
   .0696         25.      .0704       17.      .0712       16.
   .0720         18.      .0728       17.      .0736       14.
   .0744         13.      .0752       15.      .0760       14.
   .0768         13.      .0776       16.      .0784       15.
   .0792         11.      .08         11.
 15    1.        10.0      10.0        6.5      1.6      0
 13   10.
936
2649
3619
4271
5863
6987
```

INSTRUMENTATION AND NAVIGATION BASELINE...

15 2 7 101 2485

0.0000	72.8	.0008	80.4	.0016	66.8
.0024	43.2	.0032	36.5	.0040	35.5
.0048	35.4	.0056	34.2	.0064	51.2
.0072	64.4	.0080	58.8	.0088	66.4
.0096	95.2	.0104	120.	.0112	110.
.0120	76.	.0128	61.6	.0136	62.4
.0144	54.8	.0152	57.2	.0160	66.4
.0168	64.8	.0176	60.8	.0184	60.8
.0192	61.2	.0200	67.2	.0208	77.6
.0216	72.8	.0224	55.6	.0232	48.4
.0240	45.2	.0248	37.4	.0256	36.3
.0264	39.5	.0272	40.4	.0280	40.0
.0288	35.5	.0296	26.9	.0304	24.4
.0312	25.7	.0320	29.6	.0328	35.9
.0336	32.1	.0344	25.	.0352	24.1
.0360	26.2	.0368	30.8	.0376	31.
.0384	22.8	.0392	16.8	.0400	16.9
.0408	19.7	.0416	22.1	.0424	25.4
.0432	26.2	.0440	20.7	.0448	18.5
.0456	21.2	.0464	19.9	.0472	15.8
.0480	14.6	.0488	14.8	.0496	15.3
.0504	14.4	.0512	13.	.0520	14.6
.0528	15.7	.0536	14.2	.0544	10.9
.0552	8.5	.0560	10.	.0568	10.2
.0576	7.7	.0584	7.7	.0592	10.1
.0600	9.9	.0608	7.2	.0616	6.2
.0624	6.2	.0632	5.5	.0640	5.6
.0648	7.7	.0656	8.8	.0664	7.5
.0672	7.	.0680	8.3	.0688	9.
.0696	7.5	.0704	5.1	.0712	4.8
.0720	5.6	.0728	5.	.0736	4.1
.0744	4.1	.0752	4.7	.0760	4.4
.0768	4.1	.0776	4.9	.0784	4.6
.0792	3.3	.08	3.3		

10 1. 10.0 10.0 6.0 1.75 1

12 10.

BASELINE SPECTRUM TRUNCATED...

```
AA         1 13983    52   269      Ø      Ø   -5.
AG         1  2090    38    55      Ø      Ø  -1Ø.
IN         1   576    12    48      Ø      Ø   -5.
AA     13989 27976    52   269      Ø      Ø   -5.
AG      2091  4142    38    54      Ø      Ø  -1Ø.
IN       577  1152    12    48      Ø      Ø   -5.
END
 743   Ø      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.
       Ø      Ø
       Ø      Ø
       Ø      Ø       Ø       Ø       Ø       Ø       Ø       Ø       Ø
       Ø      Ø       Ø       Ø       Ø.      Ø.      Ø       Ø.      Ø.
       1     55.      Ø       Ø.      Ø       Ø.      Ø.      Ø
       Ø      Ø       Ø
```

HIGH LOAD CLIPPING 90%..

```
AA         1 13938    52   269      Ø      Ø   -5.
AG         1  2090    38    55      Ø      Ø  -1Ø.
IN         1   576    12    48      Ø      Ø   -5.
AA     13989 27976    52   269      Ø      Ø   -5.
AG      2091  4142    38    54      Ø      Ø  -1Ø.
IN       577  1152    12    48      Ø      Ø   -5.
END
 743   Ø      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.
       Ø      Ø
       Ø      Ø
       Ø      Ø       Ø       Ø       Ø       Ø       Ø       Ø       Ø
       Ø      Ø       Ø       Ø       Ø.      Ø.      Ø       Ø.      Ø.
       1     55.      1      9Ø.      Ø       Ø.      Ø.      Ø
       Ø      Ø       Ø
```

ADDITION OF OVERLOADS (125%)

```
AA         1 13988    52   269      Ø      Ø   -5.
AG         1  2090    38    55      Ø      Ø  -1Ø.
IN         1   576    12    48      Ø      Ø   -5.
AA     13989 27976    52   269      Ø      Ø   -5.
AG      2091  4142    38    54      Ø      Ø  -1Ø.
IN       577  1152    12    48      Ø      Ø   -5.
END
 743   1     12.5   125.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.
       Ø      Ø
       Ø      Ø
       Ø      Ø       Ø       Ø       Ø       Ø       Ø       Ø       Ø
       Ø      Ø       Ø       Ø       Ø.      Ø.      Ø       Ø.      Ø.
       1     55.      Ø       Ø.      Ø       Ø.      Ø.      1
       Ø      Ø       Ø
```

ADDITION OF OVERLOADS (115%)

```
AA        1 13988  52    269   Ø    Ø  -5.
AG        1  2090  38     55   Ø    Ø -1Ø.
IN        1   576  12     48   Ø    Ø  -5.
AA    13989 27976  52    269   Ø    Ø  -5.
AG     2091  4142  38     54   Ø    Ø -1Ø.
IN      577  1152  12     48   Ø    Ø  -5.
END
 743   1   12.5    115.     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.
       Ø     Ø
       Ø     Ø
       Ø     Ø      Ø      Ø      Ø      Ø      Ø      Ø      Ø
       Ø     Ø      Ø      Ø      Ø.     Ø.     Ø      Ø.     Ø.
       1    55.     Ø      Ø.     Ø      Ø.     Ø.     1
       Ø     Ø      Ø
```

BASELINE MEASURED MIX...

```
AA        1 13988  52    269   Ø    Ø  -5.
AG        1  2090  38     55   Ø    Ø -1Ø.
IN        1   576  12     48   Ø    Ø  -5.
AA    13989 27976  52    269   Ø    Ø  -5.
AG     2091  4142  38     54   Ø    Ø -1Ø.
IN      577  1152  12     48   Ø    Ø  -5.
END
 743   Ø     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.
       Ø     Ø
       Ø     Ø
       Ø     Ø      Ø      Ø      Ø      Ø      Ø      Ø      Ø
       Ø     Ø      Ø      Ø      Ø.     Ø.     Ø      Ø.     Ø.
                    Ø      Ø.     Ø      Ø.     Ø.     Ø
       Ø     Ø      Ø
```

LOW LOAD TRUNCATION 70%...

```
AA        1 13988  52    269   Ø    Ø  -5.
AG        1  2090  38     55   Ø    Ø -1Ø.
IN        1   576  12     48   Ø    Ø  -5.
AA    13989 27976  52    269   Ø    Ø  -5.
AG     2091  4142  38     54   Ø    Ø -1Ø.
IN      577  1152  12     48   Ø    Ø  -5.
END
 743   Ø     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.     Ø.
       Ø     Ø
       Ø     Ø
       Ø     Ø      Ø      Ø      Ø      Ø      Ø      Ø      Ø
       Ø     Ø      Ø      Ø      Ø.     Ø.     Ø      Ø.     Ø.
       1    7Ø.     Ø      Ø.     Ø      Ø.     Ø.     Ø
       Ø     Ø      Ø
```

```
         CLIPPIN OF TENSION LOADS...
AA          1 13988    52   269    Ø     Ø   -5.
AG          1  2Ø9Ø    38    55    Ø     Ø  -1Ø.
IN          1   576    12    48    Ø     Ø   -5.
AA      13989 27976    52   269    Ø     Ø   -5.
AG       2Ø91  4142    38    54    Ø     Ø  -1Ø.
IN        577  1152    12    48    Ø     Ø   -5.
END
 743   Ø      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.
       Ø      Ø
       Ø      Ø
       Ø      Ø       Ø       Ø       Ø       Ø       Ø       Ø       Ø
       Ø      Ø       Ø       Ø       Ø.      Ø.      Ø       Ø.      Ø.
       1     55.      Ø       Ø.      1       Ø.      Ø.      Ø
       Ø      Ø       Ø

         INCREASED SEVERITY & NO. OF A TO A LOADS...
AA          1 16952    52   326    Ø     Ø   -5.
AG          1  1Ø64    38    28    Ø     Ø  -1Ø.
IN          1   288    12    24    Ø     Ø   -5.
AA      16953 33956    52   327    Ø     Ø   -5.
AG       1Ø65  2Ø9Ø    38    27    Ø     Ø  -1Ø.
IN        289   576    12    24    Ø     Ø   -5.
END
 756   Ø      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.
       Ø      Ø
       Ø      Ø
       1      Ø     327     378     7Ø6       Ø       Ø       Ø       Ø
       Ø      Ø       Ø       Ø       Ø.      Ø.      Ø       Ø.      Ø.
       1     55.      Ø       Ø.      Ø       Ø.      Ø.      Ø
       Ø      Ø       Ø

         AIR TO GROUND BASELINE...
AG          1 4242Ø    38   63Ø    4Ø   462  -1Ø.
END
1Ø92   Ø      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.      Ø.
       Ø      Ø
       Ø      Ø
       Ø      Ø       Ø       Ø       Ø       Ø       Ø       Ø       Ø
       Ø      Ø       Ø       Ø       Ø.      Ø.      Ø       Ø.      Ø.
       1     55.      Ø       Ø.      Ø       Ø.      Ø.      Ø
       Ø      Ø       Ø
```

APPENDIX B

TEST RESULTS -

SPECIMEN GEOMETRY, TEST LIMIT LOAD, LIFE

TABLE B-1. STATIC TEST RESULTS
FIBER DOMINATED LAY-UP (48/48/4)

	Specimen Number	Thickness (in.)	Width (in.)	Hole dia (in.)	Failing Load (lb)	Failing Strain μin./in.
Compression	P1-96	0.2369	1.505	0.2484	−32,850	− 9,600
	P1-97	0.2550	1.502	0.2494	−33,750	− 9,770
	P1-98	0.2624	1.506	0.2497	−34,950	−10,160
	P1-100	0.2536	1.503	0.2508	−37,500	−
	P1-101	0.2611	1.506	0.2503	−37,800	−
Tension	P1-93	0.2377	1.507	0.2492	30,400	7,670
	P1-94	0.2485	1.502	0.2493	29,150	7,040
	P1-95	0.2512	1.502	0.2501	29,300	7,060
	P1-99	0.2438	1.507	0.2514	29,400	−

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TABLE B-2. STATIC TEST RESULTS
MATRIX DOMINATED LAY-UP (16/80/4)

	Specimen Number	Thickness (in.)	Width (in.)	Hole dia (in.)	Failing Load (lb)	Failing Strain μin./in.
Compression	P2-96	0.2505	1.502	0.2504	−21,700	−12,550
	P2-97	0.2463	1.503	0.2497	−20,850	−11,820
	P2-98	0.2417	1.504	0.2502	−21,300	−13,110
	P2-100	0.2417	1.503	0.2508	−20,300	−
	P2-101	0.2568	1.506	0.2501	−21,100	−
	P3-18	0.2714	1.506	0.2498	−21,580	−
	P3-43	0.2737	1.506	0.2494	−23,750	−
Tension	P2-93	0.2327	1.504	0.2470	14,700	7,440
	P2-94	0.2416	1.501	0.2500	15,950	7,910
	P2-95	0.2451	1.503	0.2500	16,000	7,840
	P2-99	0.2434	1.504	0.2507	16,100	−

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TABLE B-3. CONSTANT AMPLITUDE FATIGUE TEST RESULTS FIBER DOMINATED LAY-UP (48/48/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole dia (in.)	Max Load (lb)	Cycles to Failures
P1-1	0.2577	1.514	0.2498	25,390	1,970
P1-25	0.2602	1.498	0.2504		1,705
P1-51	0.2645	1.502	0.2511		1,310
P1-69	0.2571	1.502	0.2521		1,670
P1-89	0.2550	1.503	0.2527		1,690
P1-4	0.2643	1.517	0.2628	22,000	5,940
P1-61	0.2599	1.510	0.2512		6,260
P1-80	0.2642	1.512	0.2512		5,020
P1-26	0.2638	1.498	0.2511	16,930	157,260
P1-28	0.2636	1.498	0.2513		140,780
P1-91	0.2672	1.504	0.2512		186,900

Stress Ratio R = −1.0

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TABLE B-4. CONSTANT AMPLITUDE FATIGUE TEST RESULTS MATRIX DOMINATED LAY-UP (16/80/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole dia (in.)	Max Load (lb)	Cycles to Failures
P2-1	0.2521	1.522	0.2505	14,800	450
P2-39	0.2558	1.505	0.2499		250
P2-80	0.2520	1.507	0.2501		330
P2-22	0.2603	1.502	0.2520	12,820	750
P2-52	0.2451	1.504	0.2535		510
P2-81	0.2480	1.510	0.25[illegible]		800
P2-4	0.2524	1.505	0.2523	9,870	10,840
P2-19	0.2556	1.506	0.2506		7,990
P2-26	0.2595	1.500	0.2525		8,010
P2-28	0.2546	1.505	0.2532		7,990
P2-36	0.2534	1.497	0.2544		8,300
P2-48	0.2488	1.501	0.2513		5,650
P2-62	0.2570	1.510	0.2531		10,920
P2-66	0.2574	1.503	0.2526		9,860
P2-92	0.2496	1.505	0.2512		9,570

Stress Ratio R = −1.0

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TABLE B-5. SPECTRUM TEST RESULTS MATRIX DOMINATED LAY-UP (16/80/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole Dia (in.)	Fatigue Spectrum	100% TLL (lb)	Flight Hours to Failure
P2-14	0.2582	1.500	0.2527			1,585
P2-71	0.2572	1.502	0.2521			585
P2-76	0.2523	1.502	0.2515			187
P3-16	0.2775	1.506	0.2491		- 17,560	673
P3-21	0.2744	1.506	0.2498			704
P3-34	0.2717	1.505	0.2496			1,511
P3-39	0.2711	1.508	0.2494			718
P3-50	0.2705	1.504	0.2490			1,334
P2-24	0.2534	1.505	0.2540			6,298
P2-25	0.2466	1.497	0.2498			158,874
P2-27	0.2552	1.507	0.2510			132,075
P2-64	0.2532	1.507	0.2533			76,933
P2-90	0.2607	1.503	0.2516			88,874
P3-6	0.2736	1.507	0.2498			222,450
P3-8	0.2753	1.494	0.2498	F-15 Measured Mix-Truncated		192,966
P3-23	0.2695	1.505	0.2495		- 14,050	440.585
P3-33	0.2738	1.506	0.2495			249,874
P3-37	0.2743	1.507	0.2498			181,104
P4-4	0.2710	1.500	0.2497			28,450
P4-9	0.2670	1.499	0.2501			124,189
P4-13	0.2700	1.501	0.2498			16,875
P4-18	0.2680	1.500	0.2498			290,934
P4-27	0.2660	1.501	0.2496			304,338
P2-8	0.2533	1.501	0.2548			1,196,180
P2-30	0.2594	1.513	0.2537			837,417
P2-74	0.2597	1.503	0.2528			496,729
P3-1	0.2701	1.506	0.2500		- 11,240	1,922,982
P3-2	0.2613	1.505	0.2493			1,652,870
P3-12	0.2780	1.506	0.2495			2,793,310
P3-13	0.2797	1.505	0.2492			2,986,175
P3-26	0.2700	1.507	0.2498			2,530,135
P2-7	0.2625	1.501	0.2509			117,355
P2-23	0.2550	1.507	0.2520			4,097
P2-50	0.2493	1.501	0.2521			35,285
P2-55	0.2434	1.504	0.2532			152,448
P2-85	0.2523	1.501	0.2514	Clipping to 90% Test Limit Stress		162,616
P3-5	0.2749	1.504	0.2493			167,338
P3-28	0.2732	1.506	0.2494		- 14,050	499,495
P3-30	0.2710	1.506	0.2496			440,228
P3-35	0.2701	1.507	0.2484			95,914
P3-46	0.2743	1.505	0.2495			399,000
P2-12	0.2548	1.505	0.2517			2,000
P2-31	0.2570	1.508	0.2533	Addition of 125% Test Limit Stress Overloads		2,000
P2-33	0.2462	1.495	0.2500			10,000

GP13-0619-1

TABLE B-5. (Continued) SPECTRUM TEST RESULTS MATRIX DOMINATED LAY-UP (16/80/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole Dia (in.)	Fatigue Spectrum	100% TLL (lb)	Flight Hours to Failure
P2-35	0.2570	1.496	0.2528	Addition of 125% Test Limit Stress Overloads	− 14,050	1,000
P2-46	0.2510	1.498	0.2532			2,000
P2-53	0.2407	1.500	0.2511			19,000
P2-61	0.2503	1.510	0.2516			5,000
P2-84	0.2508	1.512	0.2538			1,000
P3-4	0.2616	1.506	0.2484			4,000
P3-11	0.2768	1.505	0.2492			24,000
P3-22	0.2696	1.506	0.2492			7,934
P3-42	0.2748	1.505	0.2497			10,703
P3-52	0 2756	1.506	0.2490			8,000
P2-56	0.2413	1.503	0.2531	Addition of Low Loads to Match the Original Measured Mix		73,694
P2-58	0.2366	1.505	0.2541			49,424
P2-59	0.2377	1.510	0.2532			55,367
P2-63	0.2572	1.507	0.2531			28,400
P2-79	0.2571	1.505	0.2516			7,391
P2-82	0.2578	1.512	0.2533			15,230
P2-89	0.2511	1.503	0.2525			46,304
P3-7	0.2734	1.504	0.2496			21,822
P3-24	0.2710	1.505	0.2499			18,707
P3-36	0.2737	1.505	0.2495			116,786
P3-38	0.2734	1.507	0.2491			16,747
P3-41	0.2763	1.506	0.2494			172,710
P4-5	0.2690	1.499	0.2498			203,317
P4-6	0.2670	1.500	0.2499			102,832
P4-10	0.2700	1.501	0.2499			204,803
P4-11	0.2710	1.501	0.2500			109,171
P4-16	0.2650	1.500	0.2499			232,338
P4-20	0.2700	1.500	0.2500			315,829
P4-21	0.2700	1.499	0.2499			53,304
P4-23	0.2710	1.501	0.2498			219,193
P4-24	0.2700	1.500	0.2503			200,457
P4-28	0.2650	1.502	0.2501			178,832
P2-11	0.2610	1.505	0.2516			19,230
P2-15	0.2590	1.508	0.2500			14,094
P2-21	0.2456	1.499	0.2510			146,565
P2-29	0.2460	1.514	0.2500			176,832
P2-40	0.2545	1.503	0.2538			9,169
P2-13	0.2480	1.498	0.2504	Truncation to 70% Test Limit Stress		226,371
P2-18	0.2587	1.503	0.2539			4,167
P2-32	0.2553	1.505	0.2534			12,631
P2-37	0.2458	1.505	0.2508			224,116
P2-41	0.2450	1.489	0.2507			300,121

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TABLE B-5. (Concluded) SPECTRUM TEST RESULTS
MATRIX DOMINATED LAY-UP (16/80/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole Dia (in.)	Fatigue Spectrum	100% TLL (lb)	Flight Hours to Failure
P2-20	0.2524	1.505	0.2511	Clipping of Tension Loads	−14,050	4,585
P2-44	0.2497	1.502	0.2530			7,704
P3-17	0.2769	1.506	0.2499			44,875
P3-20	0.2751	1.503	0.2496			26,376
P3-32	0.2742	1.504	0.2500			65,151
P3-47	0.2746	1.506	0.2491			20,204
P3-49	0.2714	1,504	0.2494			20,440
P2-9	0.2515	1.497	0.2507	Increased Severity and Number of Air-to-Air-Loads		5,474
P2-10	0.2603	1.498	0.2527			1,877
P2-45	0.2413	1.497	0.2516			2,851
P2-49	0.2413	1.499	0.2504			18,878
P2-54	0.2455	1.502	0.2533			1,905
P2-57	0.2362	1.519	0.2511			21,877
P2-69	0.2451	1.502	0.2507			2,851
P3-3	0.2636	1.506	0.2500			2,581
P3-14	0.2743	1.506	0.2493			3,298
P3-29	0.2742	1.507	0.2496			7,530
P3-45	0.2756	1.506	0.2494			5,376
P3-51	0.2723	1.502	0.2495			1,851
P2-2	0.2583	1.519	0.2537	Air-to-Ground Baseline		784,000
P2-51	0.2485	1.506	0.2526			420,139
P2-83	0.2594	1.510	0.2527			254,712
P3-9	0.2800	1.505	0.2500			1,124,317
P3-40	0.2753	1.499	0.2498			1,865,990

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TABLE B6. SPECTRUM TEST RESULTS FIBER DOMINATED LAY-UP (48/48/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole Dia. (in.)	Fatigue Spectrum	100% TLL (lb)	Flight Hours to Failure
P1-10	0.2643	1.497	0.2508	F-15 Measured Mix-Truncated	−30,100	48,729
P1-44	0.2627	1.492	0.2524			69,440
P1-57	0.2465	1.512	0.2505			6,585
P1-59	0.2496	1.522	0.2484			6,450
P1-81	0.2568	1.513	0.2522			41,151
P1-83	0.2677	1.501	0.2515			101,874
P1-87	0.2671	1.503	0.2512			91,896
P5-32	0.2720	1.498	0.2499			71,585
P5-37	0.2730	1.499	0.2502			132,874
P5-39	0.2740	1.500	0.2501			89,704
P5-46	0.2710	1.501	0.2497			60,329
P5-54	0.2680	1.499	0.2500			32,874
P1-11	0.2670	1.500	0.2503		−28,330	227,370
P1-14	0.2658	1.497	0.2500			211,450
P1-16	0.2623	1.497	0.2519			2,700
P1-63	0.2685	1.511	0.2521			119,585
P1-73	0.2662	1.503	0.2536			2,930
P1-85	0.2582	1.503	0.2523			151,054
P1-22	0.2660	1.501	0.2462		−26,560	371,930
P1-23	0.2646	1.505	0.2501			372,151
P1-41	0.2611	1.498	0.2507			253,704
P1-77	0.2557	1.503	0.2518			230,870
P1-79	0.2646	1.509	0.2513			406,440
P1-13	0.2596	1.497	0.2507	Clipping to 90% Test Limit Stress	−28,330	198,417
P1-24	0.2634	1.499	0.2530			187,410
P1-27	0.2635	1.502	0.2499			229,780
P1-38	0.2662	1.498	0.2503			282,104
P1-90	0.2647	1.504	0.2527			230,706
P1-17	0.2594	1.510	0.2505	Addition of 115% Test Limit Stress Overloads		20,000
P1-29	0.2602	1.508	0.2503			1,700
P1-32	0.2631	1.513	0.2528			2,000
P1-47	0.2653	1.502	0.2510			24,585
P1-49	0.2580	1.499	0.2482			1,000
P1-68	0.2630	1.502	0.2510			8,030
P1-78	0.2650	1.506	0.2540			1,000
P1-82	0.2660	1.515	0.2530			2,000
P1-6	0.2647	1.496	0.2513	Addition of Low Leads to Match the Original Measured Mix	−30,100	93,910
P1-46	0.2681	1.498	0.2512			49,822
P1-84	0.2646	1.500	0.2520			1,852
P1-88	0.2638	1.505	0.2523			54,338
P5-29	0.2720	1.499	0.2500			3,369
P5-33	0.2710	1.501	0.2503			42,551
P5-34	0.2680	1.500	0.2501			5,169
P5-38	0.2730	1.498	0.2503			39,882

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EFFECT OF FIGHTER ATTACK SPECTRUM ON COMPOSITE FATIGUE LIFE.(U)

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TABLE B-6. (Continued) SPECTRUM TEST RESULTS FIBER DOMINATED LAY-UP (48/48/4)

Specimen Number	Thickness (in.)	Width (in.)	Hole Dia (in.)	Fatigue Spectrum	100% TLL (lb)	Flight Hours to Failure
P5-44	0.2700	1.501	0.2495	Addition of Low Loads to Match the Original Measured Mix	−30,100	111,304
P5-48	0.2690	1.500	0.2501			12,550
P5-49	0.2700	1.500	0.2494			58,716
P5-51	0.2750	1.500	0.2499			87,160
P5-52	0.2730	1.501	0.2498			60,449
P5-56	0.2630	1.498	0.2498			9,832
P1-8	0.2654	1.496	0.2530		−28,330	161,400
P1-40	0.2626	1.504	0.2516			65,581
P1-66	0.2649	1.503	0.2542			51,073
IR-16	0.2762	1.507	0.2507			107,251
IR-17	0.2765	1.504	0.2494			144,638
P1-18	0.2660	1.511	0.2521	Truncation to 70% Test Limit Stress	−28,330	336,099
P1-19	0.2655	1.515	0.2496			365,156
P1-21	0.2606	1.501	0.2503			118,653
P1-34	0.2647	1.495	0.2523			229,930
P1-43	0.2653	1.501	0.2493			88,333
P1-52	0.2577	1.499	0.2540			9,032
P1-15	0.2668	1.500	0.2505			80,205
P1-33	0.2593	1.496	0.2507	Clipping of Tension Loads		151,440
P1-35	0.2637	1.499	0.2502			5,440
P1-36	0.2625	1.496	0.2520			18,870
IR-18	0.2782	1.505	0.2495			168,450
IR-19	0.2787	1.506	0.2501			209,032
P1-64	0.2632	1.512	0.2528			293,704
P1-37	0.2595	1.500	0.2496			15,850
P1-42	0.2670	1.497	0.2505	Increased Severity and Number of Air-to-Air Loads	−28,330	850
P1-45	0.2581	1.498	0.2512			10,870
P1-50	0.2639	1.499	0.2502			580
P1-67	0.2651	1.503	0.2513			1,720
P1-70	0.2661	1.502	0.2523			300
IR-22	0.2772	1.506	0.2497			4,851
IR-25	0.2665	1.505	0.2497			7,655
IR-26	0.2662	1.507	0.2501			715
P1-12	0.2650	1.497	0.2525	Air-to-Ground Baseline		883,099
P1-31	0.2656	1.505	0.2496			1,486,190
P1-53	0.2541	1.497	0.2506			344,490
P1-55	0.2606	1.502	0.2513			331,920
P1-86	0.2663	1.503	0.2523			653,405
IR-21	0.2757	1.504	0.2496			598,362
IR-20	0.2736	1.510	0.2496			119,145

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APPENDIX C

WEIBULL STATISTICAL ANALYSES

OF DATA

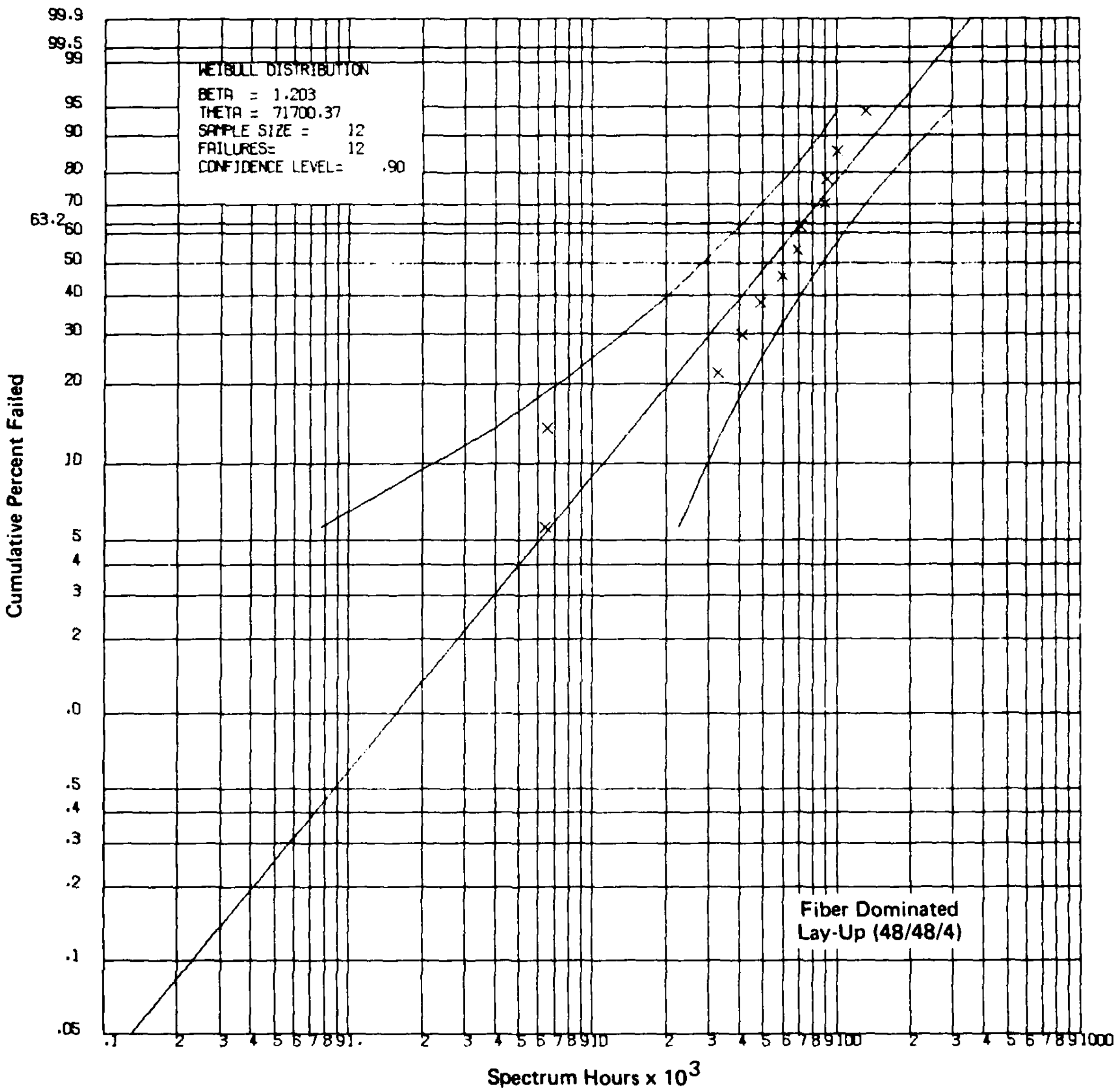

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Figure C-1. F-15 Measured Mix-Truncated

TLL = 90% F_{CU}

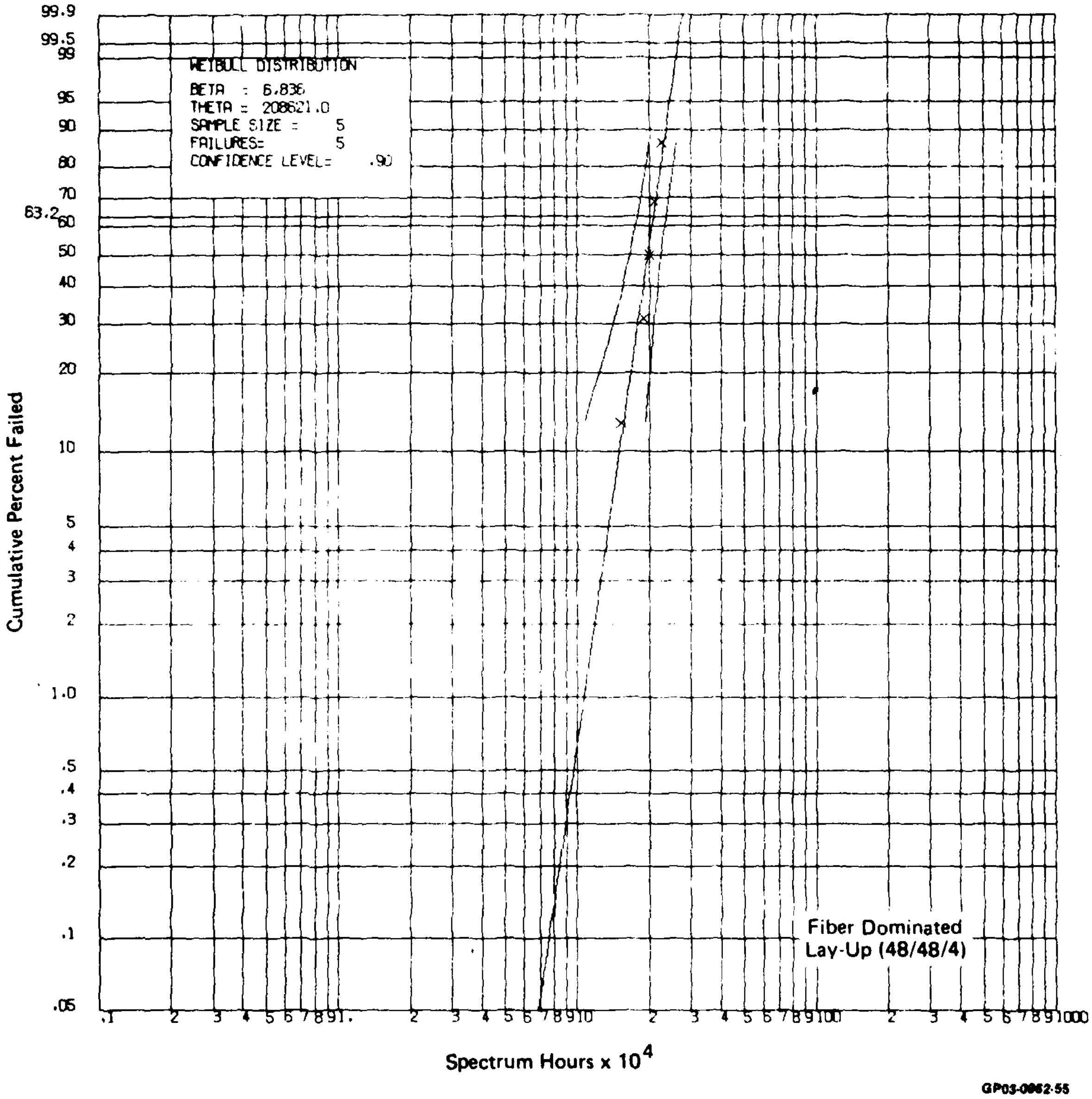

Figure C-2. F-15 Measured Mix-Truncated

TLL = 85% F_{cu}

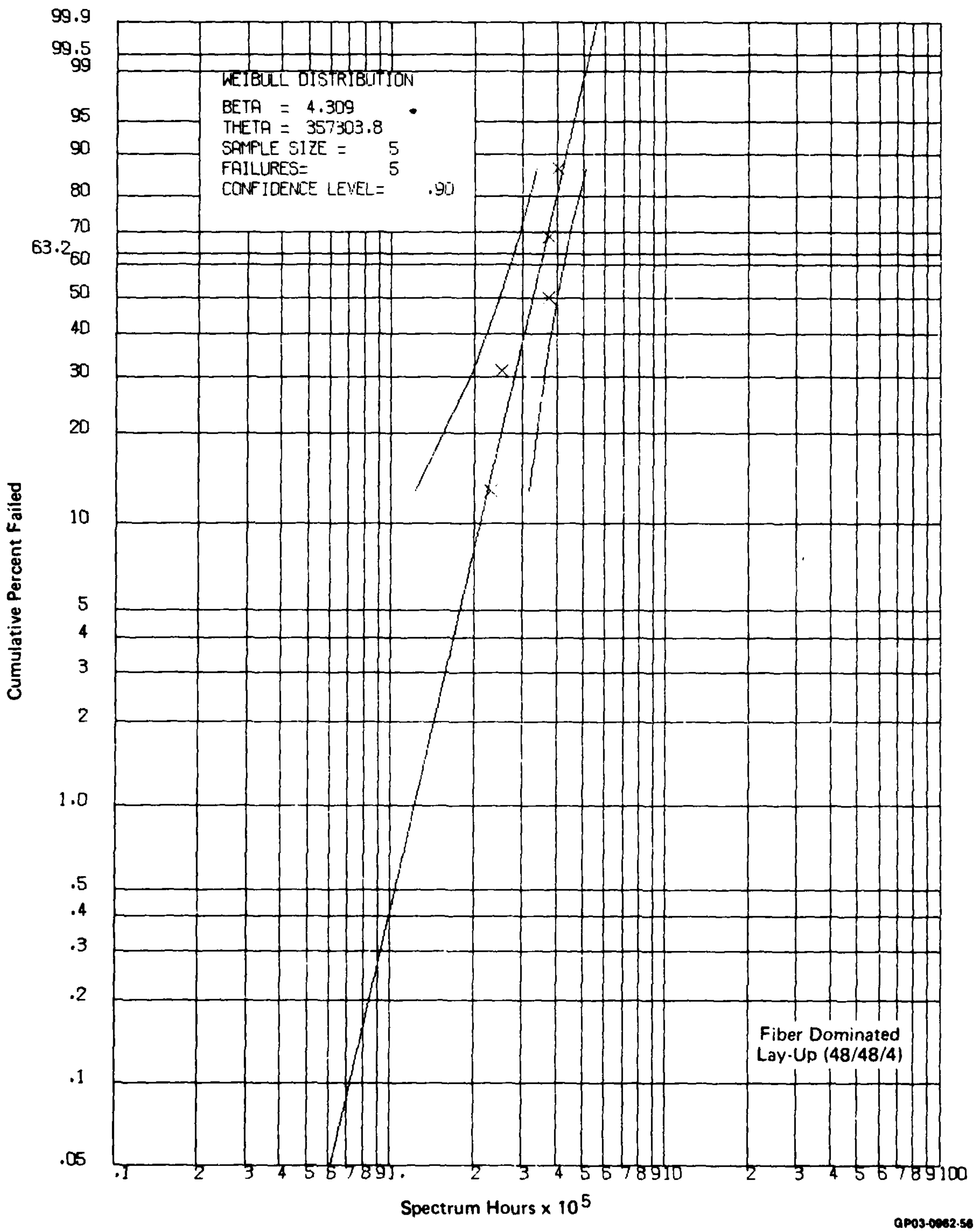

Figure C-3. F-15 Measured Mix-Truncated
TLL = 80% F_{cu}

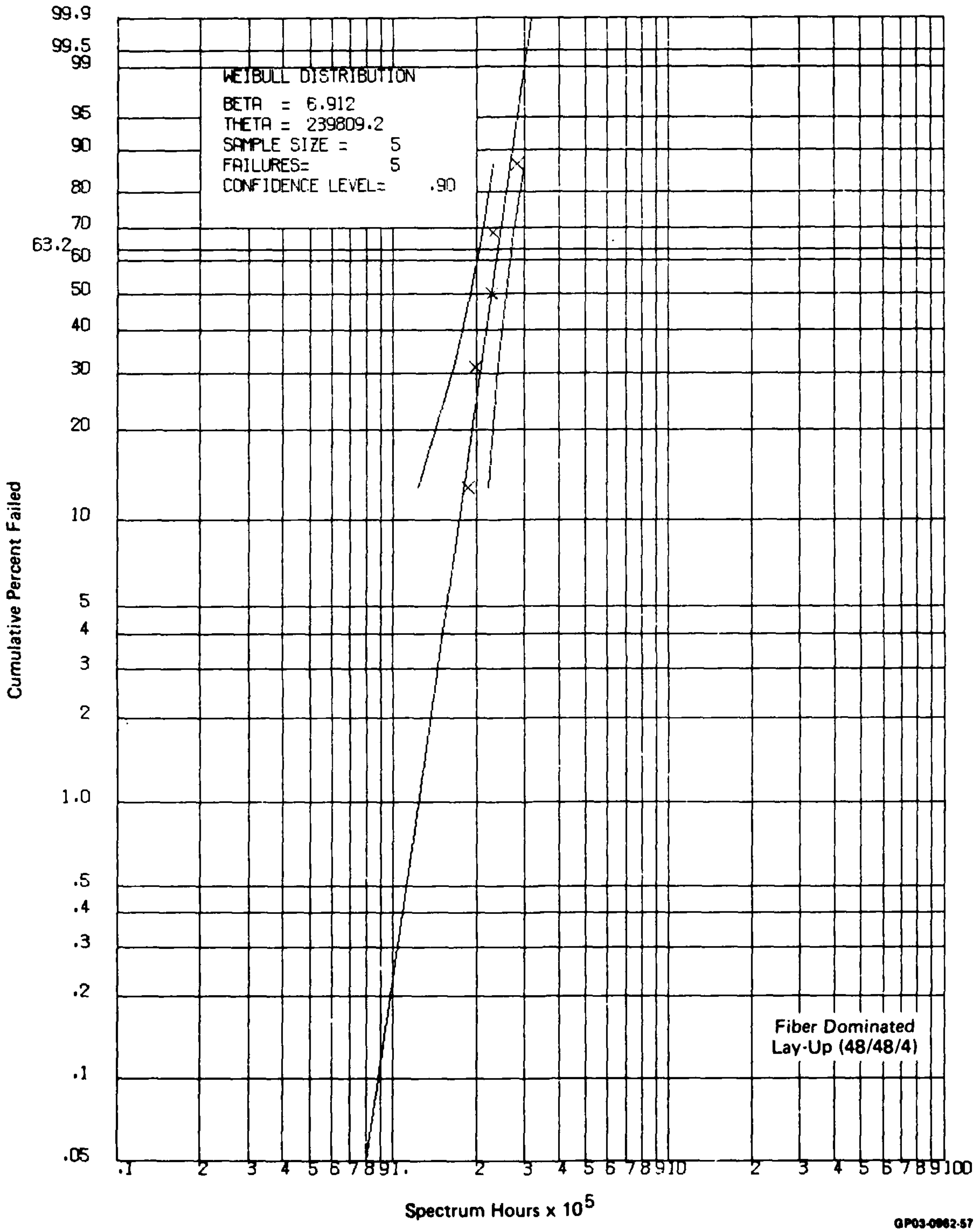

Figure C-4. Clipping to 90% Test Limit Stress

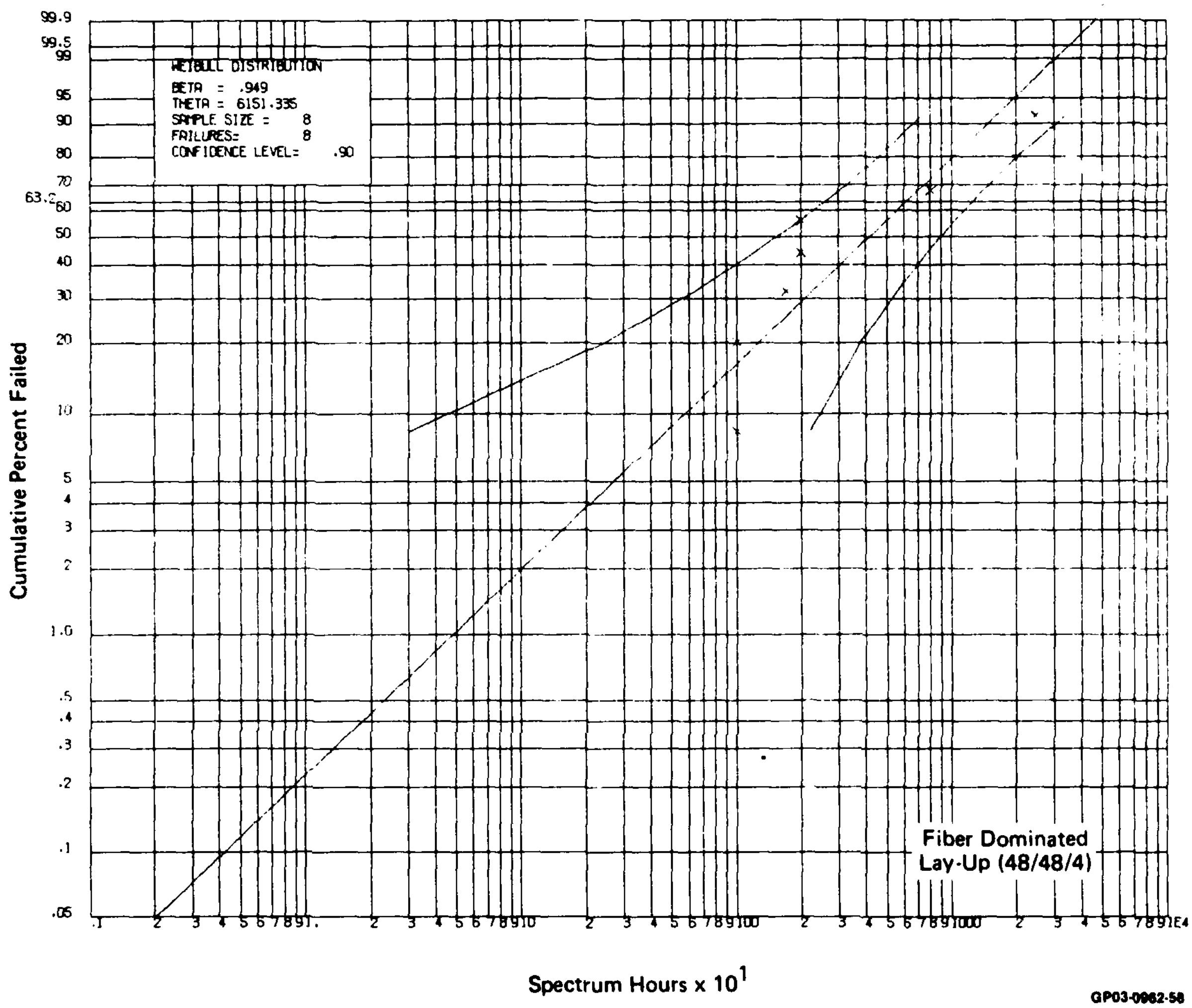

Figure C-5. Addition of 115% Test Limit Stress Overloads

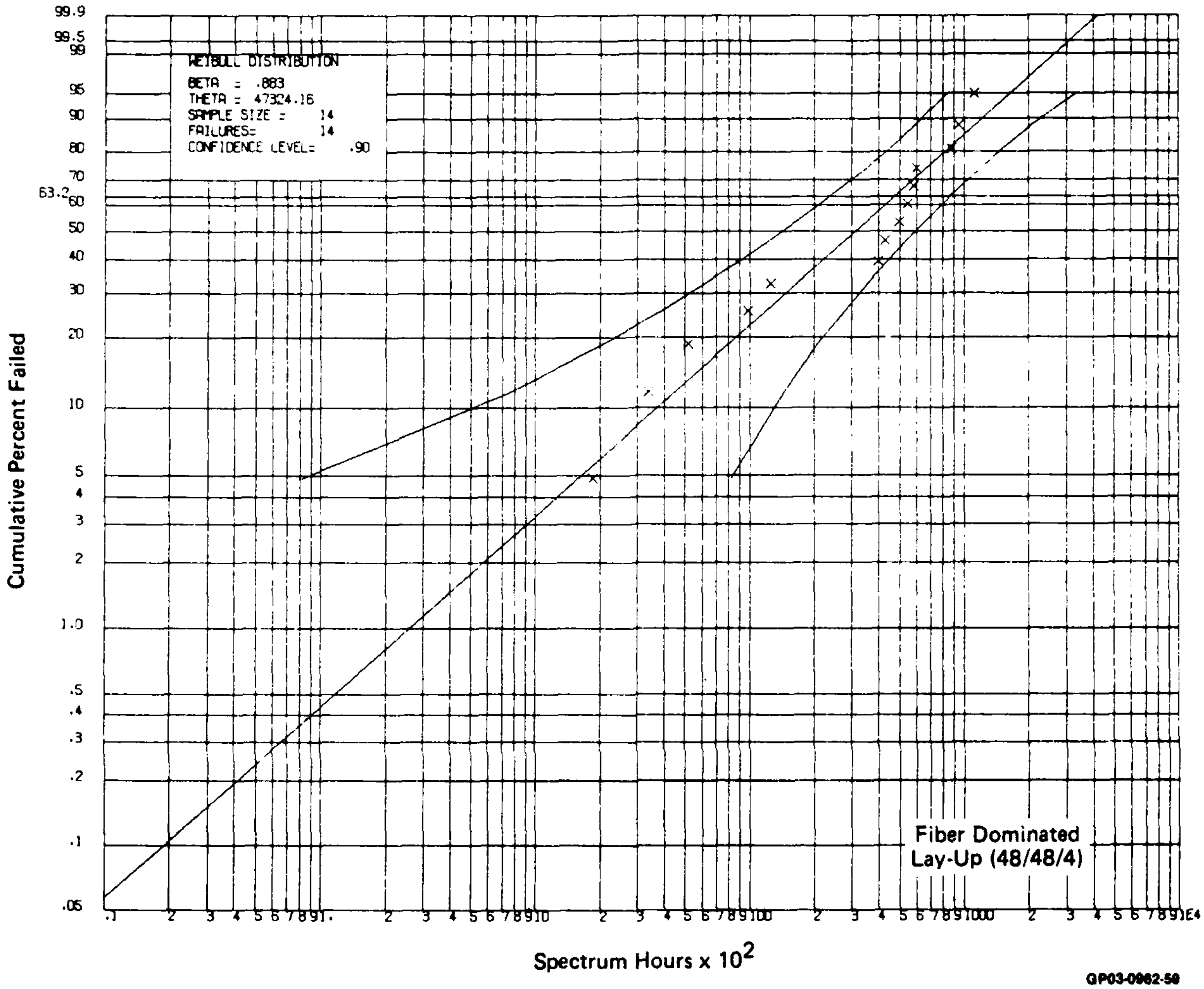

Figure C-6. Addition of Loads to Match Original Measured Mix
TLL = 90% F_{cu}

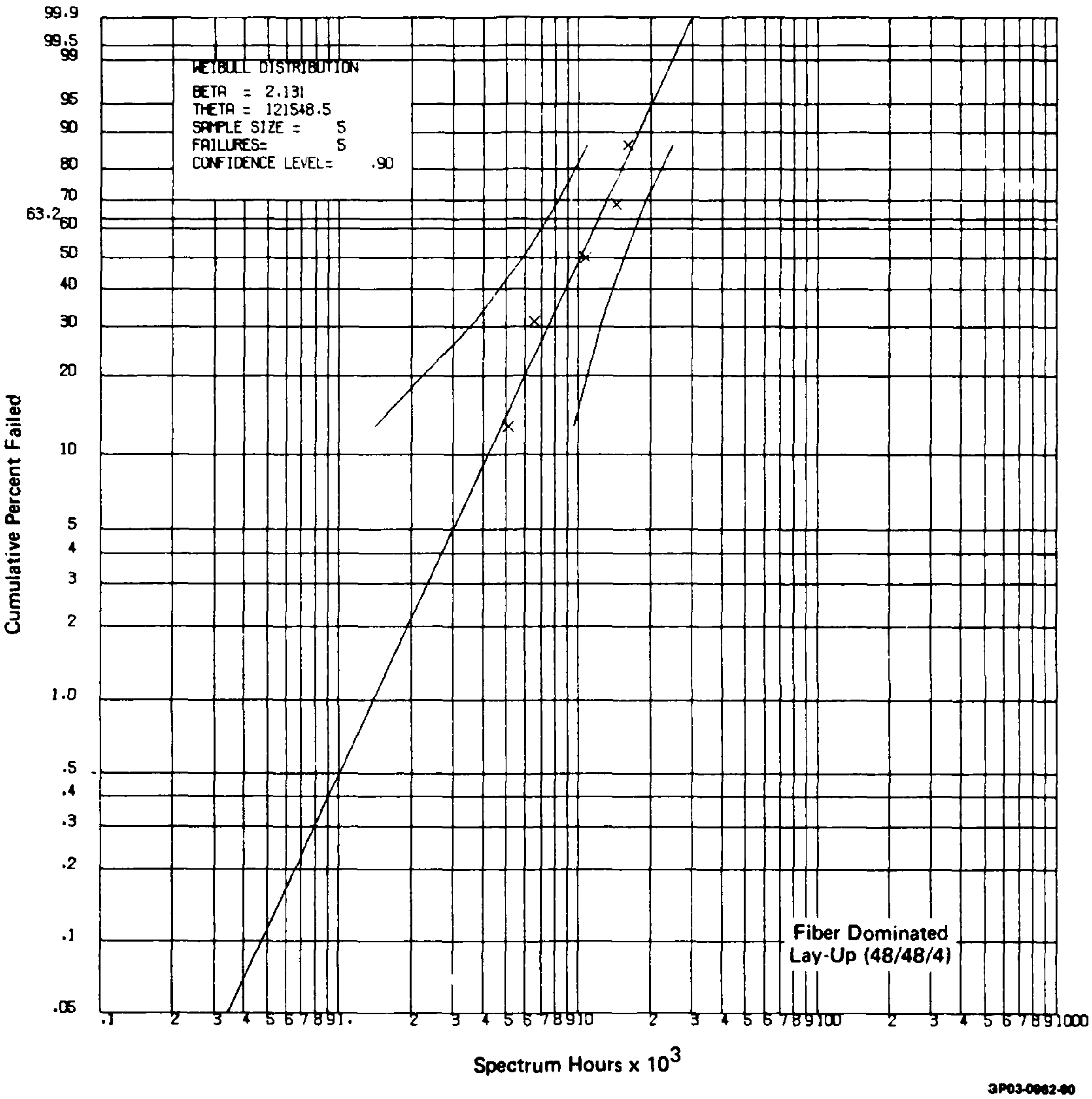

Figure C-7. Addition of Loads to Match Original Measured Mix

TLL = 85% F_{cu}

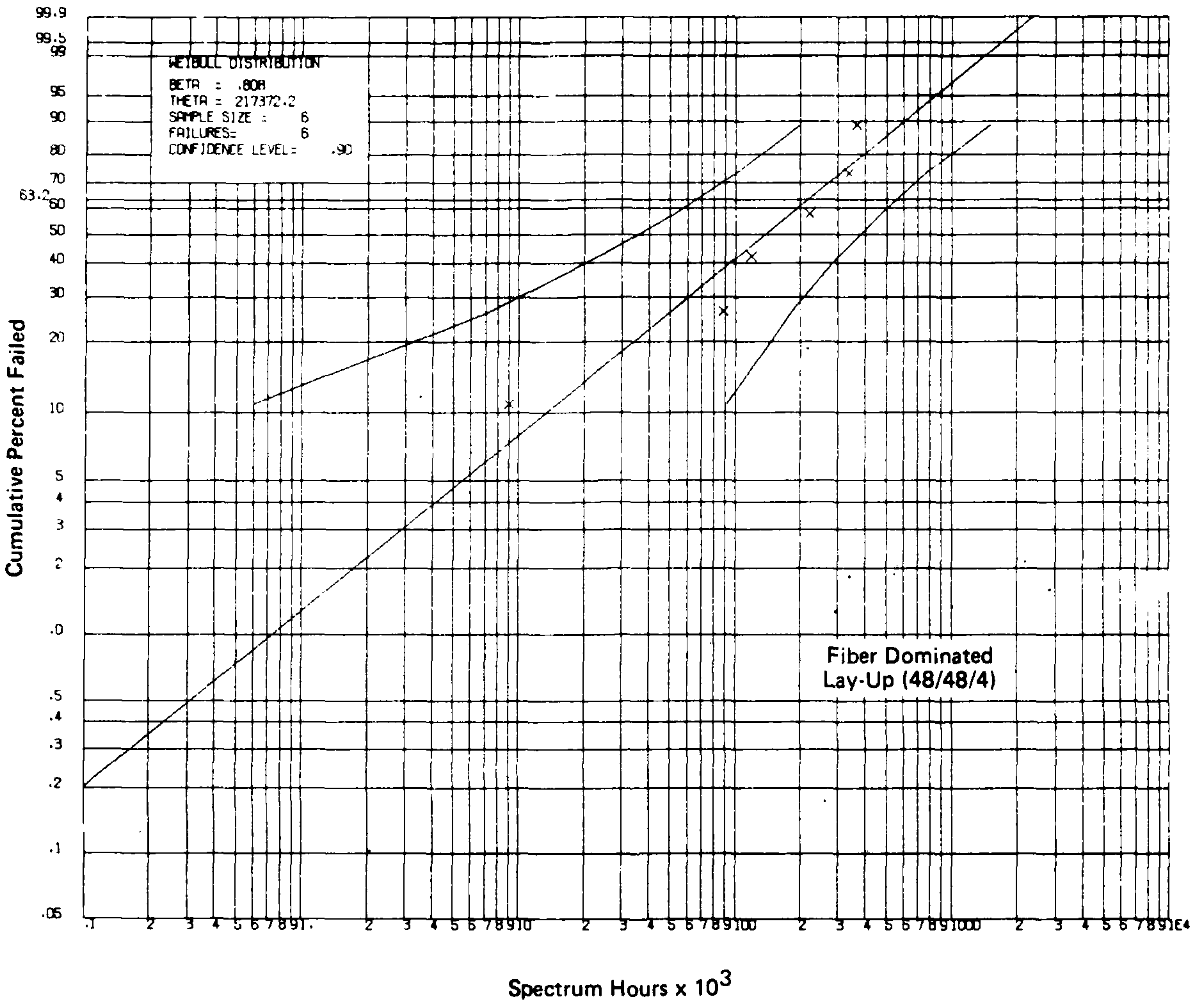

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Figure C-8. Truncation to 70% Test Limit Stress

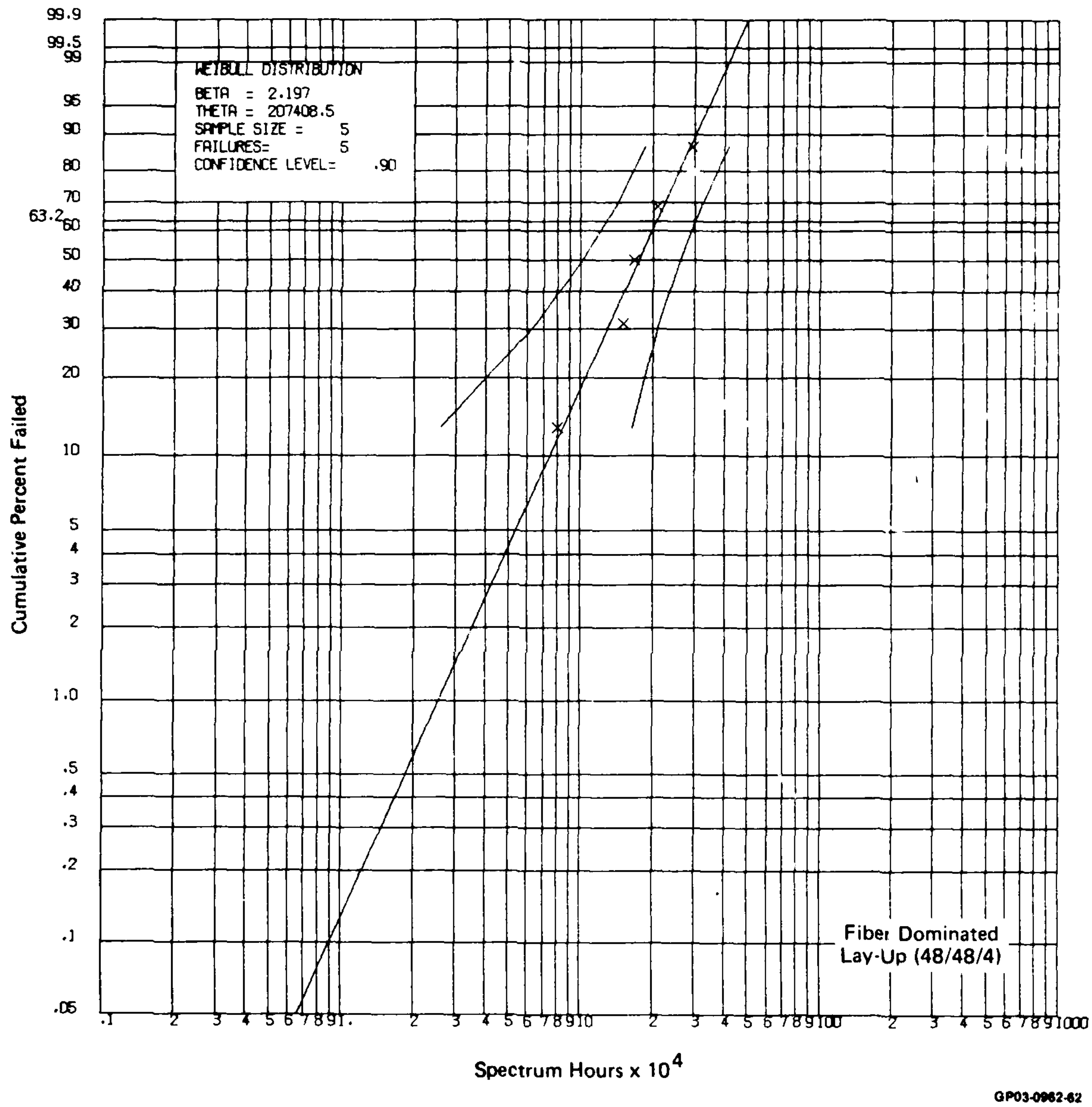

Figure C-9. Clipping of Tension Loads

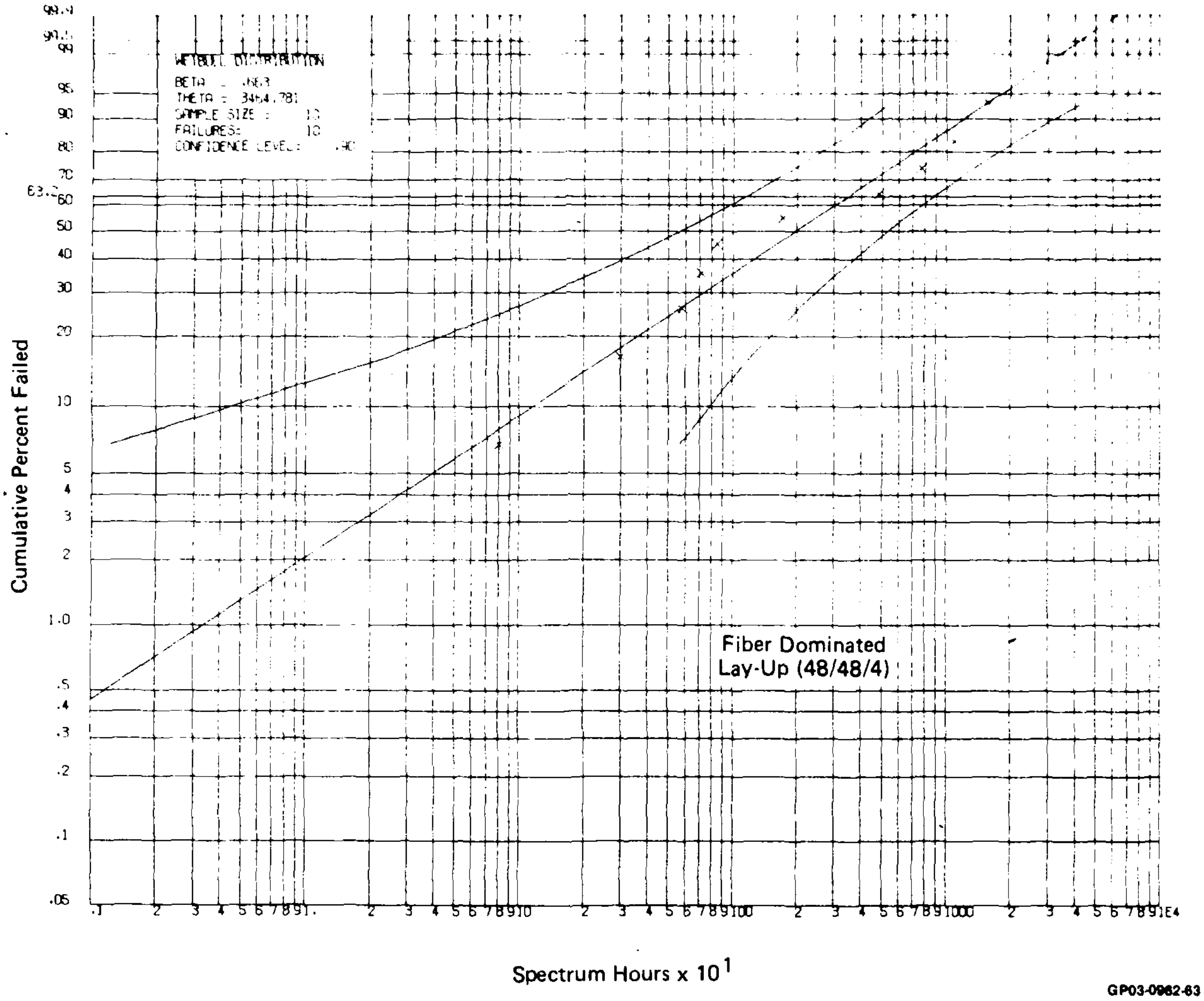

Figure C-10. Increased Severity and Number of Air-to-Air Loads

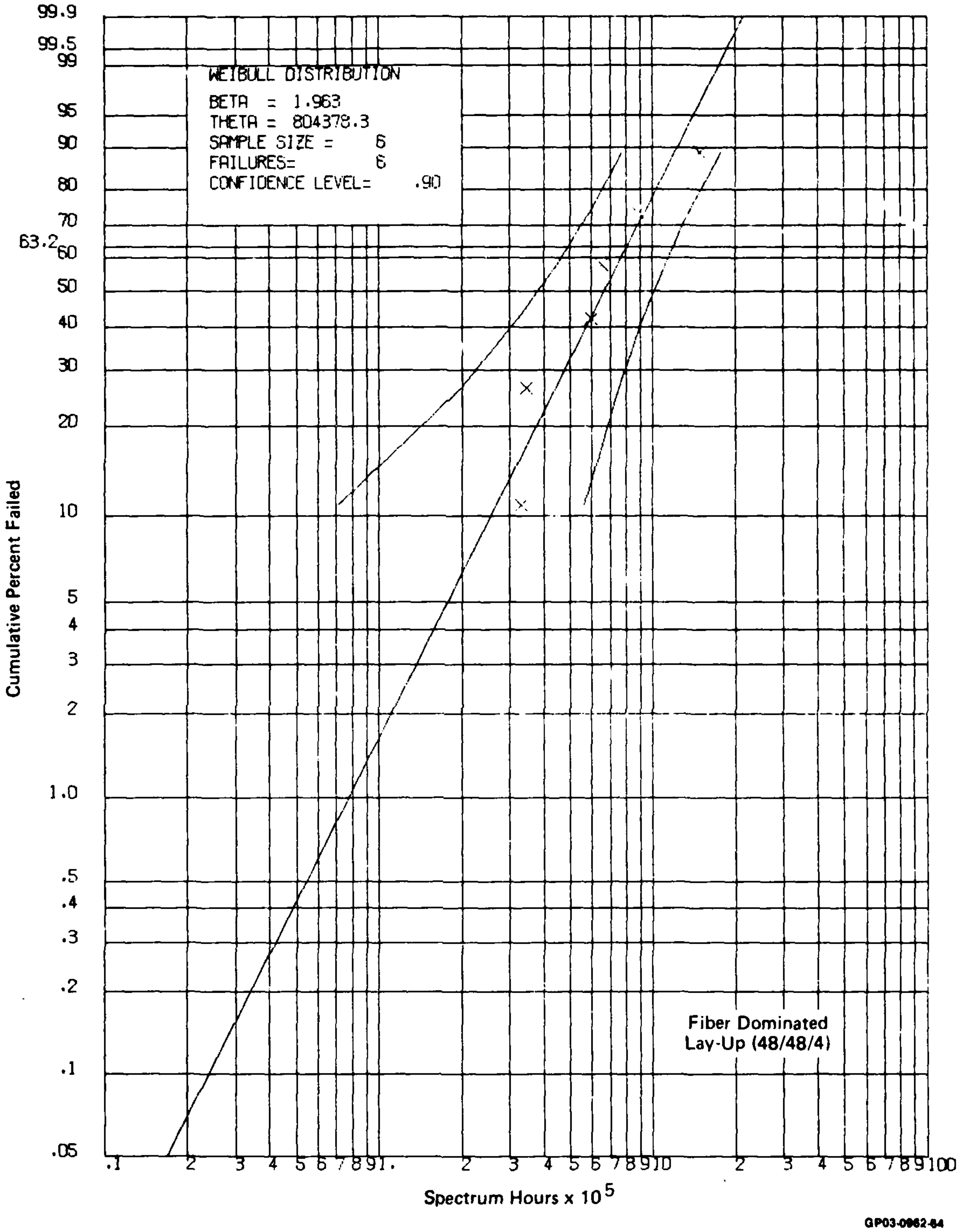

Figure C-11. Air-to-Ground Baseline

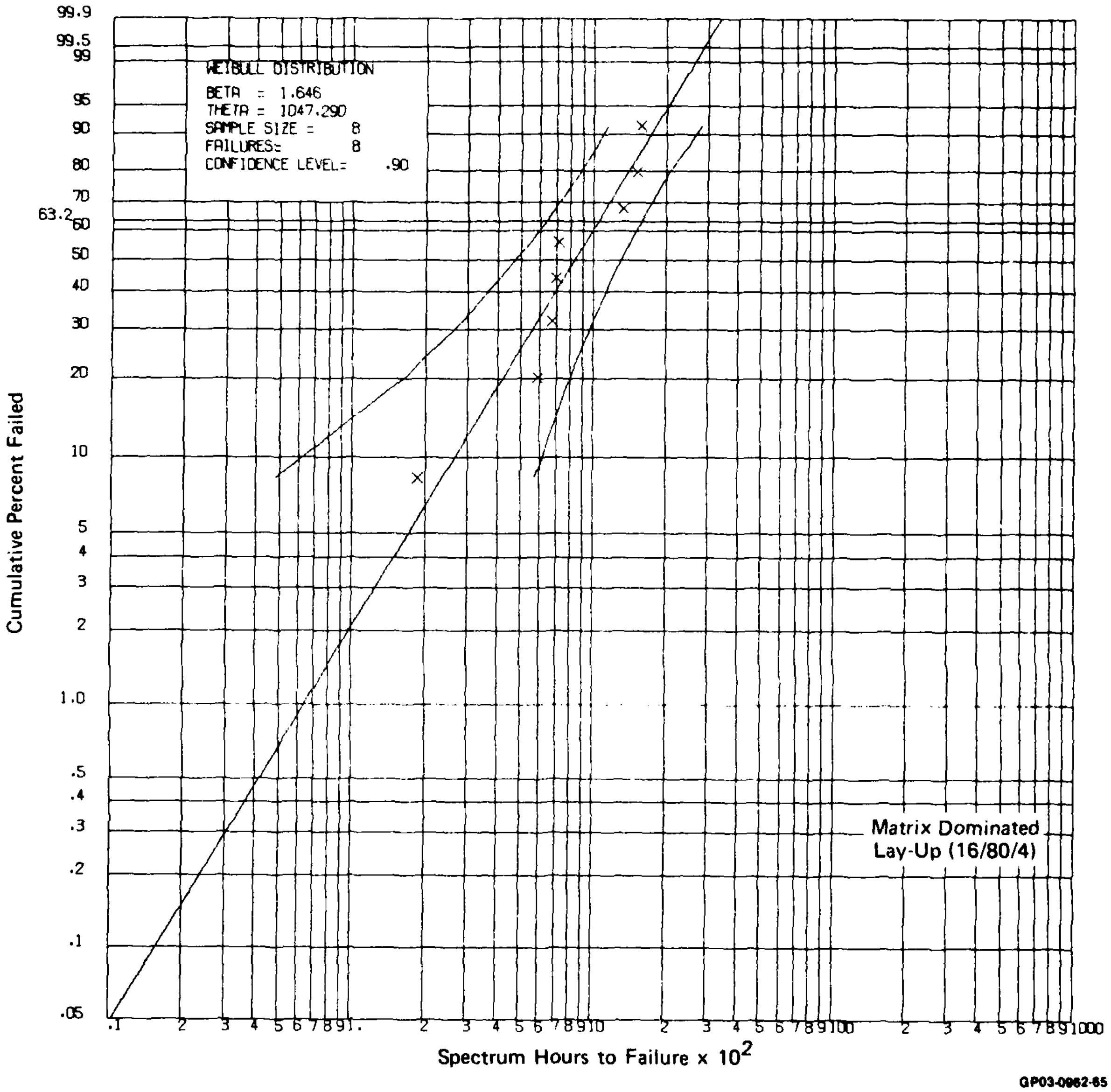

Figure C-12. F-15 Measured Mix-Truncated
TLL = 85% F_{cu}

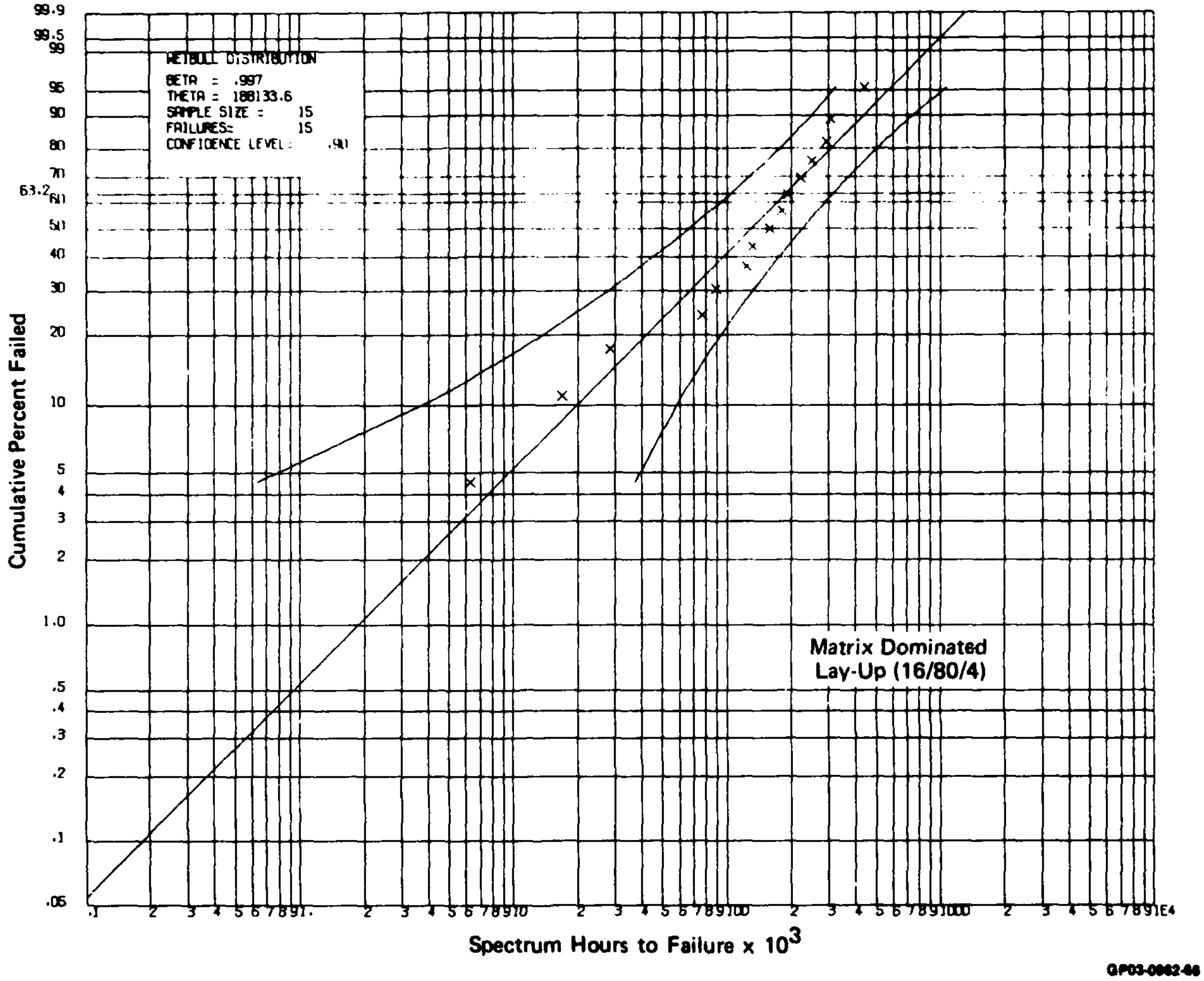

Figure C-13. F-15 Measured Mix-Truncated
TLL = 66% F_{CU}

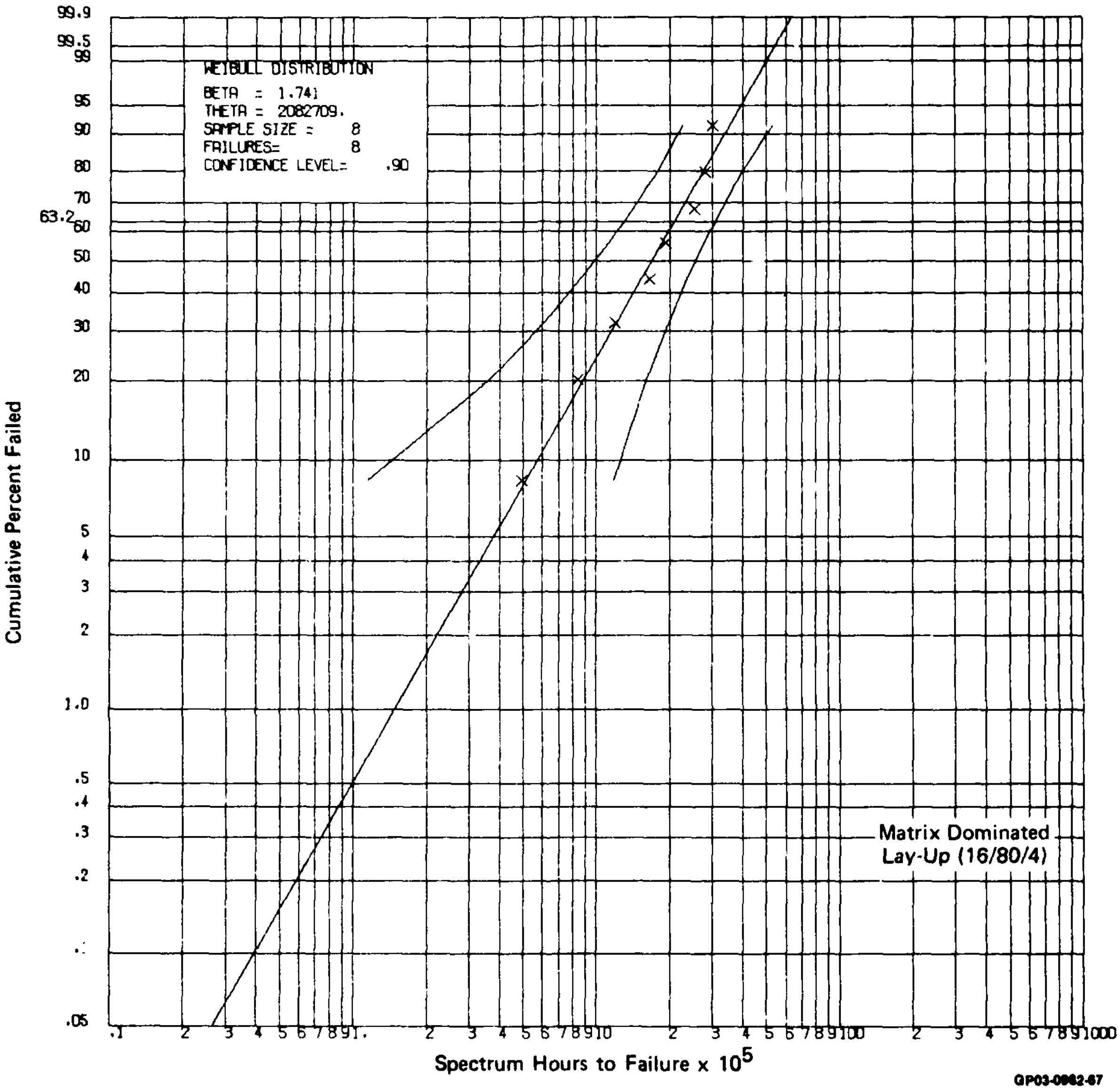

Figure C-14. F-15 Measured Mix-Truncated
TLL = 55% F_{cu}

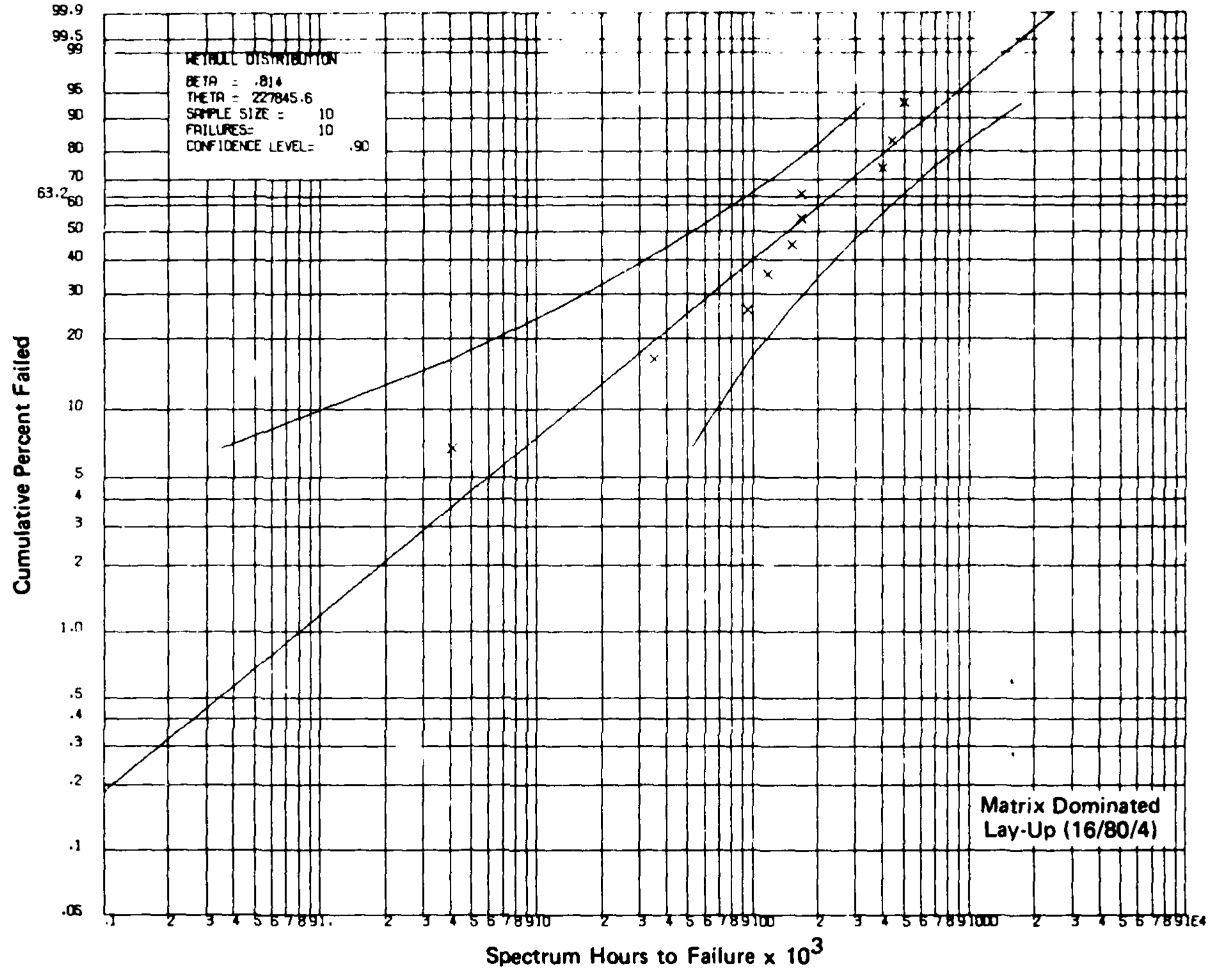

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Figure C-15. Clipping to 90% Test Limit Stress.

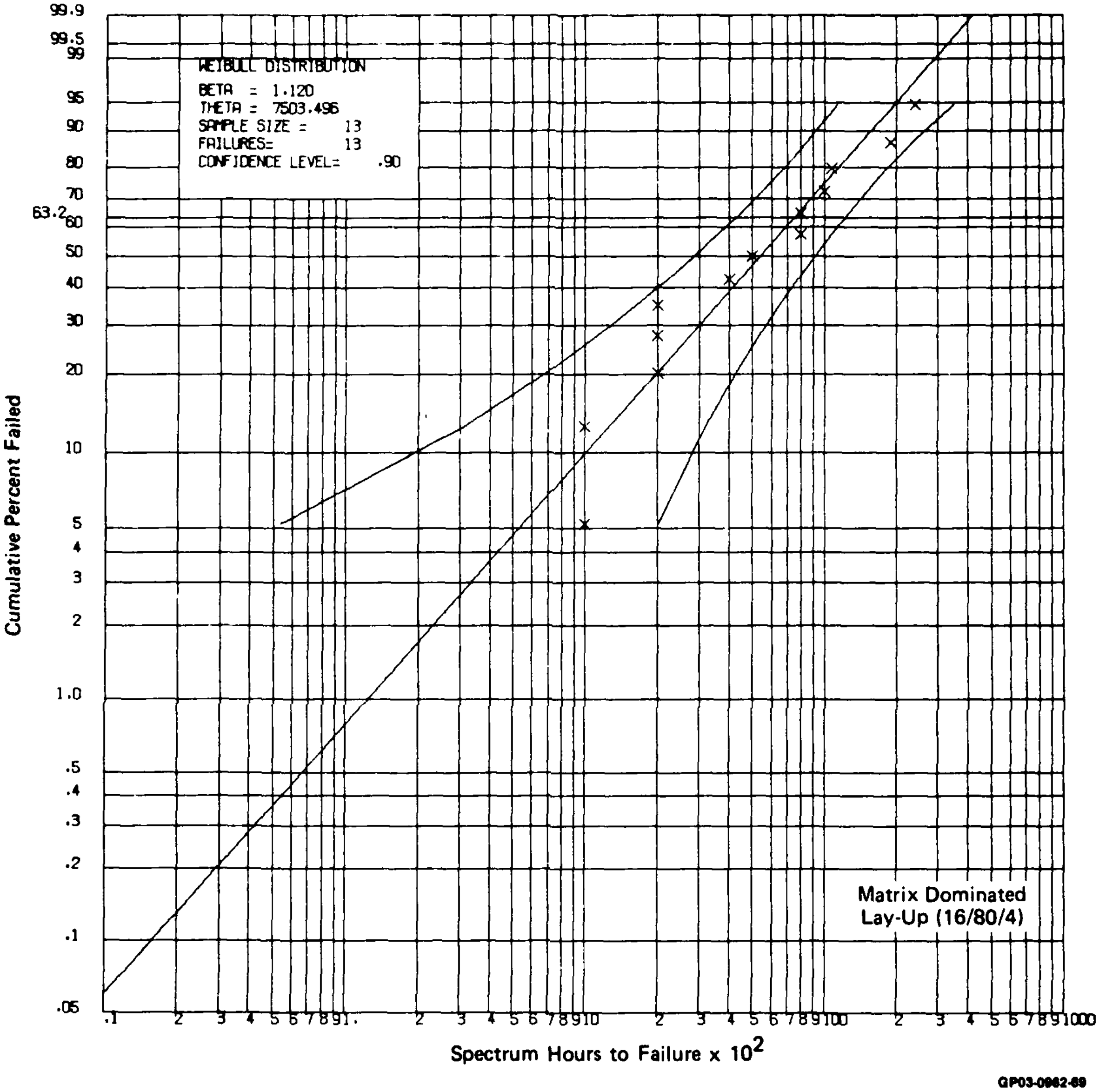

Figure C-16. Addition of 125% Test Limit Stress Overloads

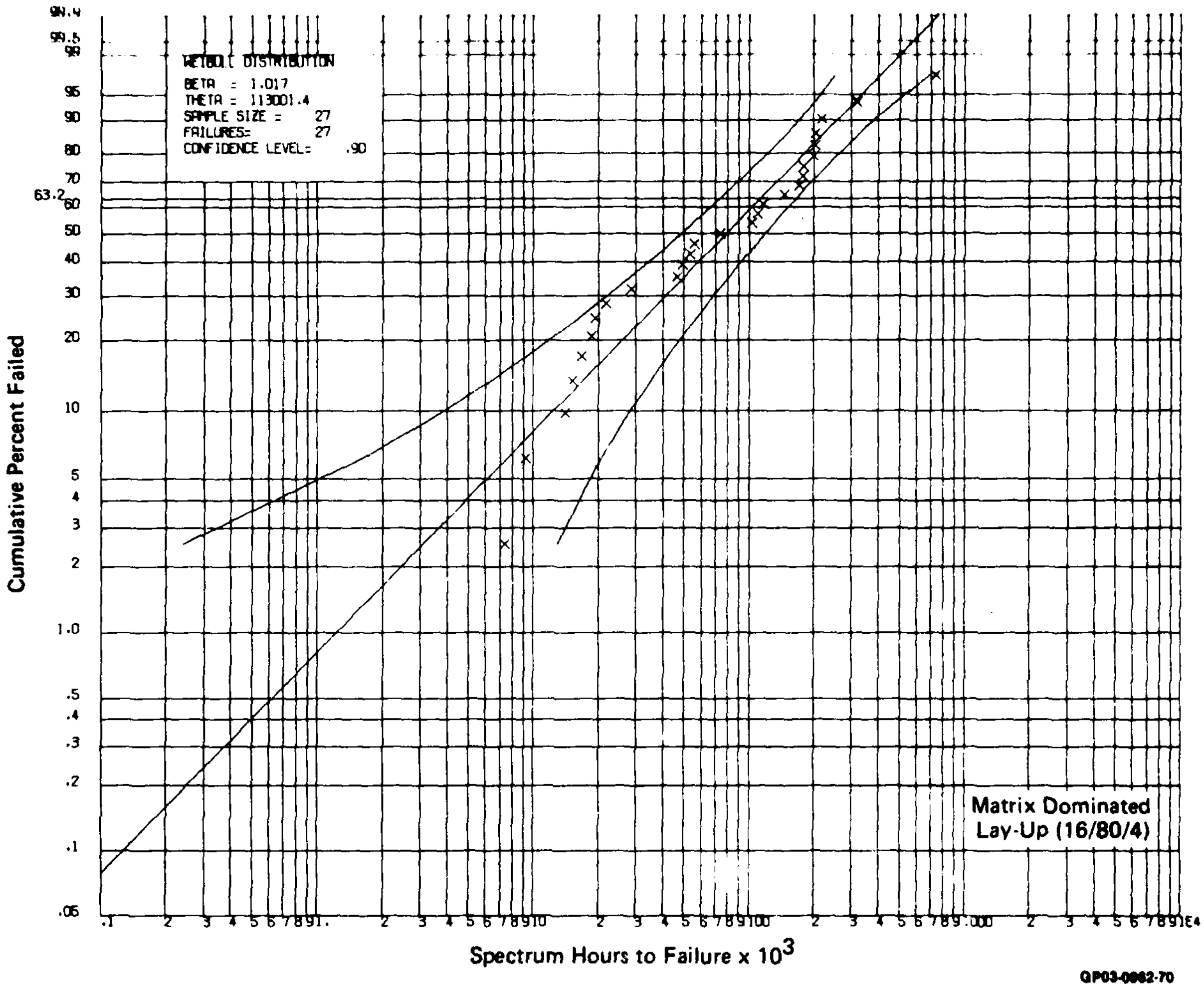

Figure C-17. Addition of Low Loads to Match Original Measured Mix

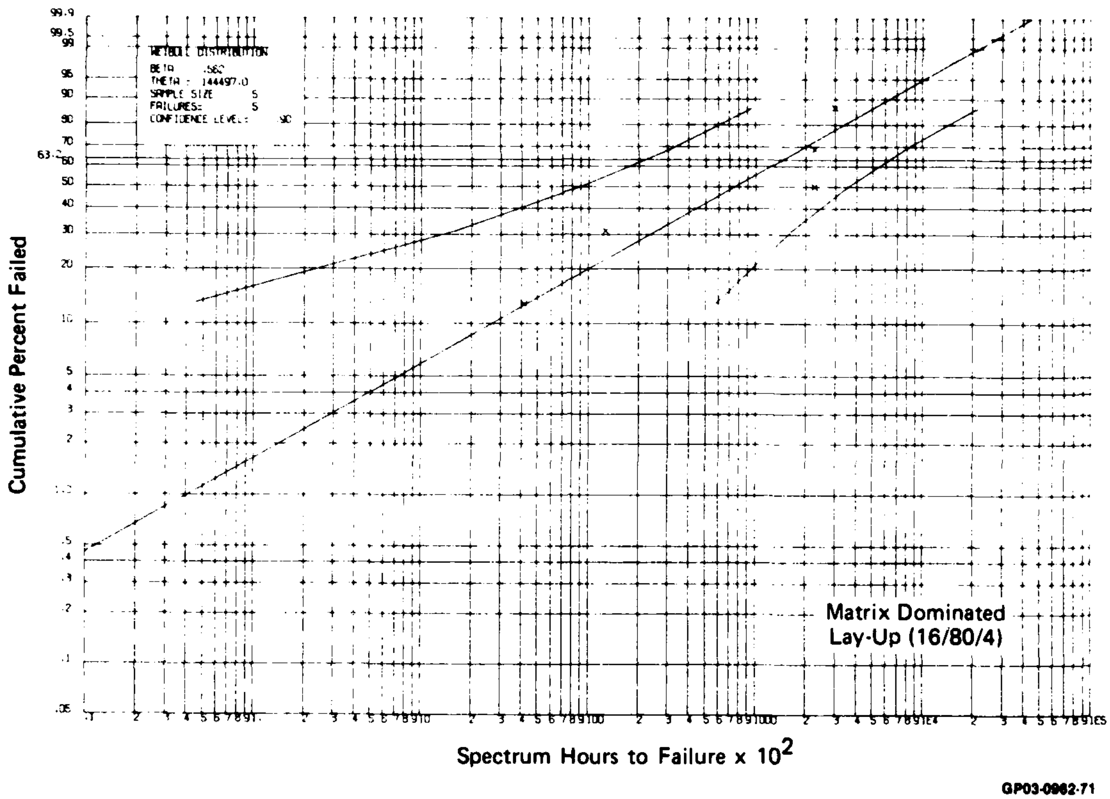

Figure C-18. Truncation to 70% Test Limit Stress

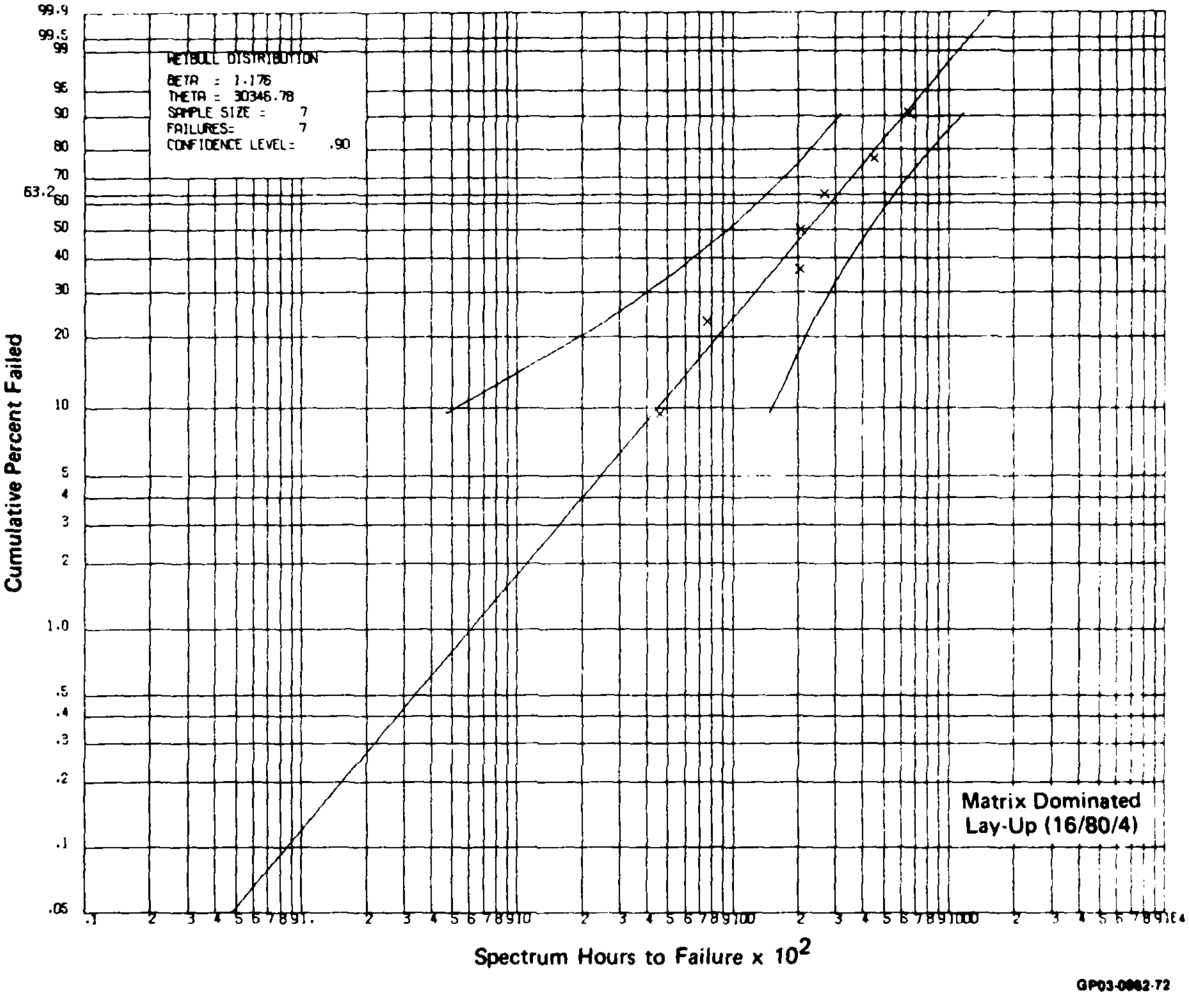

Figure C-19. Clipping of Tension Loads

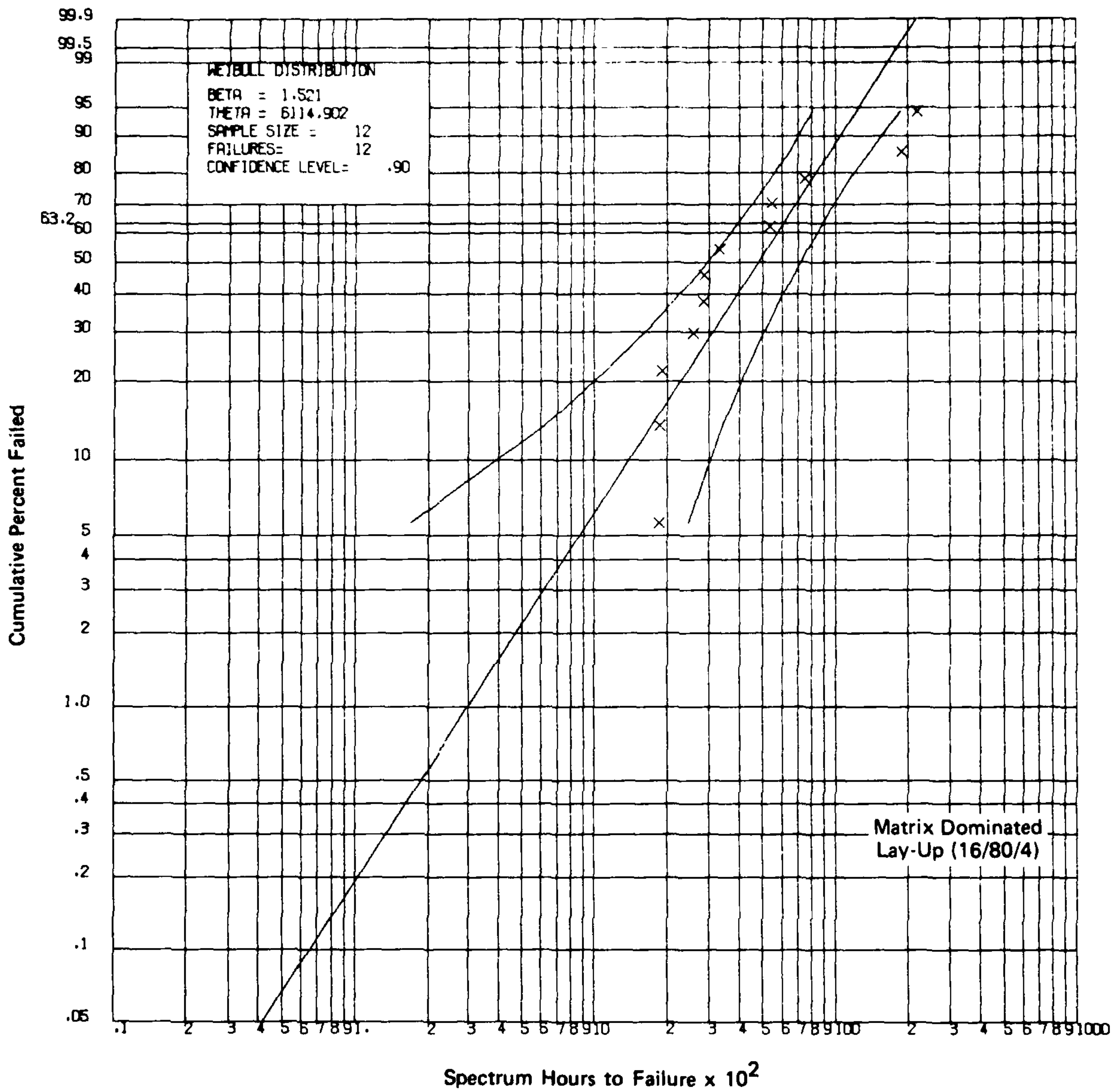

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Figure C-20. Increased Severity and Number of Air-to-Air Loads

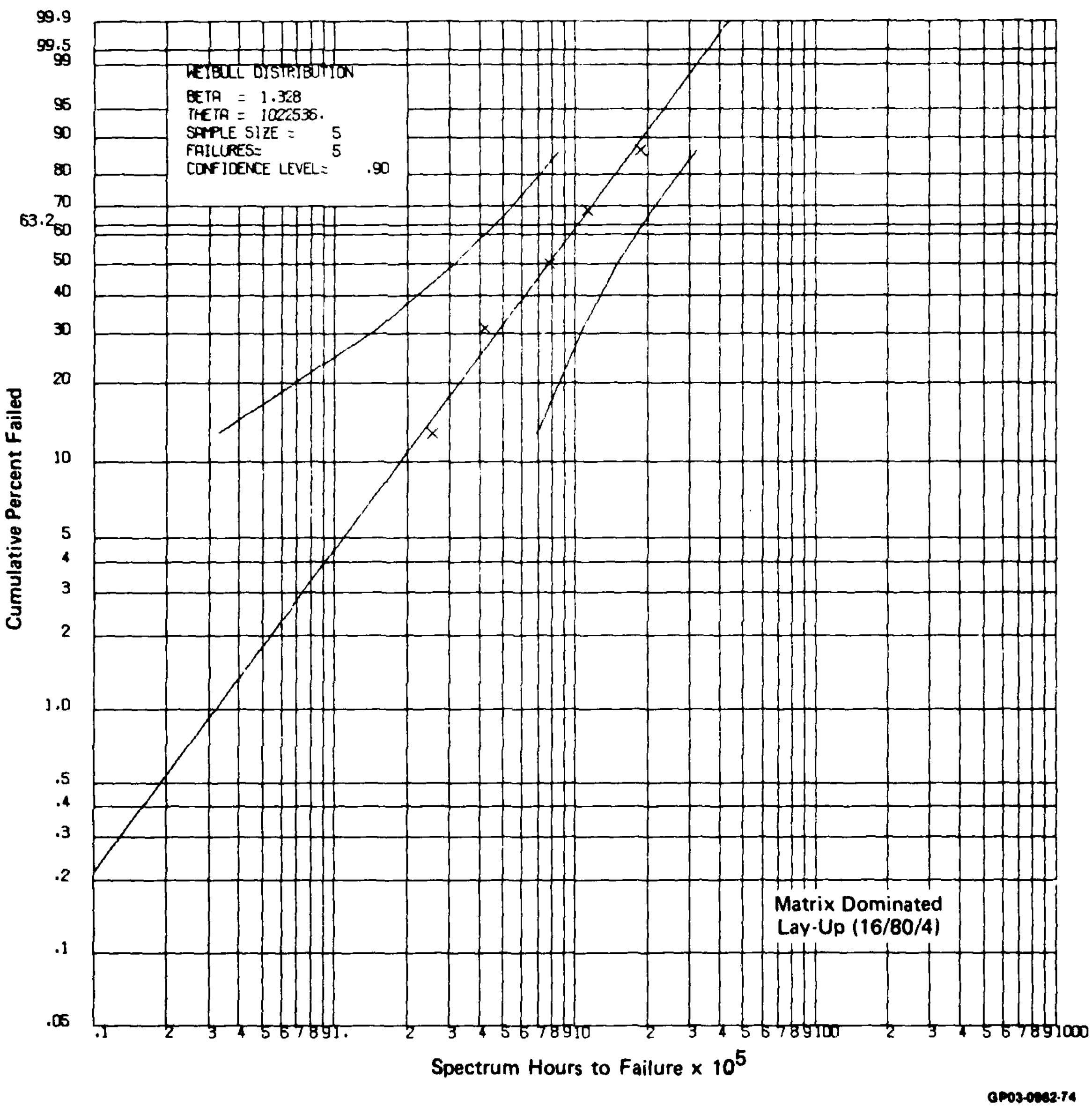

Figure C-21. Air-to-Ground Baseline

APPENDIX D

PANEL CONFIGURATION

P1-57	P1-58	P1-59	P1-60
P1-53	P1-54	P1-55	P1-56
P1-49	P1-50	P1-51	P1-52
P1-45	P1-46	P1-47	P1-48
P1-41	P1-42	P1-43	P1-44
P1-37	P1-38	P1-39	P1-40
P1-33	P1-34	P1-35	P1-36
P1-29	P1-30	P1-31	P1-32
P1-25	P1-26	P1-27	P1-28
P1-21	P1-22	P1-23	P1-24
P1-17	P1-18	P1-19	P1-20
P1-13	P1-14	P1-15	P1-16
P1-9	P1-10	P1-11	P1-12
P1-5	P1-6	P1-7	P1-8
P1-1	P1-2	P1-3	P1-4
P1-61	P1-62	P1-63	P1-64
P1-65	P1-66	P1-67	P1-68
P1-69	P1-70	P1-71	P1-72
P1-73	P1-74	P1-75	P1-76
P1-77	P1-78	P1-79	P1-80
P1-81	P1-82	P1-83	P1-84
P1-85	P1-86	P1-87	P1-88
P1-89	P1-90	P1-91	P1-92

P1-95	P1-98	P1-101
P1-94	P1-97	P1-100
P1-93	P1-96	P1-99

GP13-0619-2

Figure D-1. Panel 1 Specimen Location
Fiber Dominated Lay-Up 48/48/4

P2-57	P2-58	P2-59	P2-60
P2-53	P2-54	P2-55	P2-56
P2-49	P2-50	P2-51	P2-52
P2-45	P2-46	P2-47	P2-48
P2-41	P2-42	P2-43	P2-44
P2-37	P2-38	P2-39	P2-40
P2-33	P2-34	P2-35	P2-36
P2-29	P2-30	P2-31	P2-32
P2-25	P2-26	P2-27	P2-28
P2-21	P2-22	P2-23	P2-24
P2-17	P2-18	P2-19	P2-20
P2-13	P2-14	P2-15	P2-16
P2-9	P2-10	P2-11	P2-12
P2-5	P2-6	P2-7	P2-8
P2-1	P2-2	P2-3	P2-4
P2-89	P2-90	P2-91	P2-64
P2-85	P2-86	P2-87	P2-68
P2-81	P2-82	P2-83	P2-72
P2-77	P2-78	P2-79	P2-76
P2-73	P2-74	P2-75	P2-80
P2-69	P2-70	P2-71	P2-84
P2-65	P2-66	P2-67	P2-88
P2-61	P2-62	P2-63	P2-92

P2-96	P2-101	P2-95
P2-97	P2-100	P2-94
P2-98	P2-99	P2-93

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Figure D-2. Panel 2 Specimen Location Matrix Dominated Lay-Up 16/80/4

P3-1	P3-2	P3-3	P3-4
P3-5	P3-6	P3-7	P3-8
P3-9	P3-10	P3-11	P3-12
P3-13	P3-14	P3-15	P3-16
P3-17	P3-18	P3-19	P3-20
P3-21	P3-22	P3-23	P3-24
P3-25	P3-26	P3-27	P3-28
P3-29	P3-30	P3-31	P3-32
P3-33	P3-34	P3-35	P3-36
P3-37	P3-38	P3-39	P3-40
P3-41	P3-42	P3-43	P3-44
P3-45	P3-46	P3-47	P3-48
P3-49	P3-50	P3-51	P3-52

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Figure D-3. Panel 3 Specimen Location Matrix Dominated Lay-Up 16/80/4

IR-16	IR-20	IR-24
IR-17	IR-21	IR-25
IR-18	IR-22	IR-26
IR-19	IR-23	IR-27

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Figure D-4. Panel 4 Specimen Location Fiber Dominated Lay-Up 48/48/4

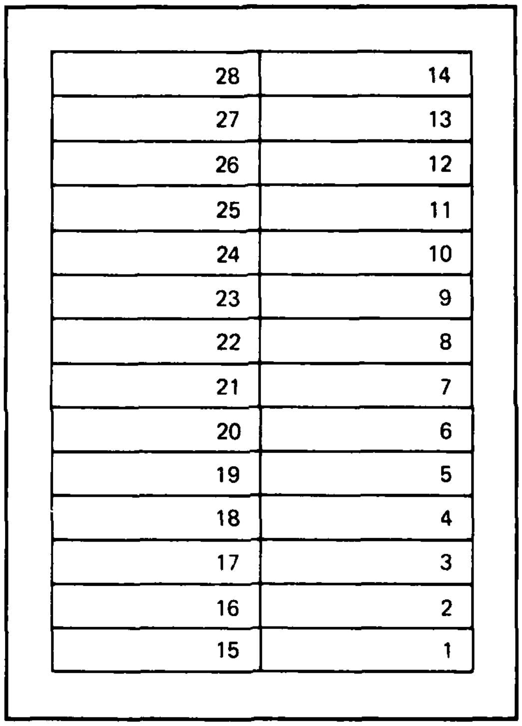

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Figure D-5. Panel 5 Specimen Location Matrix Dominated Lay-Up 16/80/4

56	42
55	41
54	40
53	39
52	38
51	37
50	36
49	35
48	34
47	33
46	32
45	31
44	30
43	29

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Figure D-6. Panel 6 Specimen Location Fiber Dominated Lay-Up 48/48/4